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RESEARCH MEMORANDUM

A FLIGHT INVESTIGATION OF THE HANDLING CHARACTERISTICS OF A FIGHTER AIRPLANE CONTROLLED THROUGH AN ATTITUDE TYPE OF AUTOMATIC PILOT

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NATIONAL ADVISORY COMMITTEE
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SUMMARY

A flight investigation was made to obtain experimental information on the handling qualities of a fighter airplane which a human pilot controlled by supplying signals to an attitude type of automatic pilot. An automatic-pilot control stick which simulated a conventional type of control stick was used to introduce signals into the automatic pilot.

In maneuvering with the attitude automatic pilot, the pilots much preferred the control-force characteristics provided by a damper feel system to those provided by a spring feel system. In general, the pilots did not consider the attitude type of control system to be as desirable for rapid maneuvering (such as required in air-to-air gunnery) as a conventional type of control system. For flight operations involving little or no maneuvering and when flying in rough air, the airplane attitude and heading stabilization greatly improved the flying qualities of the airplane. For precision flying, such as tracking a nonmaneuvering or a mildly maneuvering target and in strafing runs, the pilot was able to do about equally well when using either the attitude control having the damper feel system or the conventional control system.

INTRODUCTION

In the past automatic pilots have been used in airplanes mainly to provide airplane heading and attitude stabilization and/or to provide increased damping to the airframe. In general they have not been used for rapid airplane maneuvering. Recently there has been considerable interest in the concept of making the automatic pilot a part of the maneuvering control system of the airplane and having the human pilot control and maneuver the airplane by supplying signals to the automatic pilot (see ref. 1). This interest stems from the potential possibilities afforded the control system designer for achieving a more desirable and uniform airplane response to control applications by the pilots. With control systems of this type the stability and response characteristics of the airplane can be considerably less dependent on the airplane flight condition than with conventional control systems. Also, depending upon the type of automatic pilot used, the response resulting from the input signal by the human pilot can be varied. For example, a given pilot's input signal might produce a proportional change in attitude angle, angular rate, or acceleration.

A need exists for futher information on the advantages and disadvantages of the various control schemes made possible through use of automatic systems. In order to obtain experimental information on the flying qualities of an airplane controlled through automatic pilots, the NACA is conducting a flight program using various types of automatic pilots. This paper presents results obtained in a flight investigation of an attitude type of automatic-pilot control system which was installed in a fighter-type airplane.

When the human pilot is a part of the airplane control system, his impressions of the airplane handling qualities and his ability to perform precision flight are influenced not only by the response and damping characteristics of the airplane—automatic-pilot combination but also by the automatic-pilot controller characteristics. Because of this an appreciable part of the flight program was concerned with the automatic-pilot controller characteristics.

Some of the contents of this paper have been published previously in reference 2.

SYMBOLS

a_n normal acceleration, g units

a_y lateral acceleration, g units

F_{c_l} automatic-pilot control force, lateral, lb

F_{c_p} automatic-pilot control force, fore and aft, lb

h_p pressure altitude, ft

K_f servo feedback gain, volts per radian δ_s

K_{β} pendulum gain, volts/g

K_{θ} pitch vertical gyro gain, volts/radian

$K_{\dot{\theta}}$ pitch rate gyro gain, volts/radian/sec

K_{ϕ} roll vertical gyro gain, volts/radian

$K_{\dot{\phi}}$ roll rate gyro gain, volts/radian/sec

K_{ψ} directional gyro gain, volts/radian

$K_{\dot{\psi}}$ yaw rate gyro gain, volts/radian/sec

M Mach number

p rolling velocity, radian/sec

q pitching velocity, radian/sec

r yawing velocity, radian/sec

R servo system input signal, volts (used in ground tests)

V_i indicated airspeed, knots

α angle of attack, deg

β angle of sideslip, deg

δ_{a_T} total aileron deflection, deg

δ_{c_l} automatic-pilot control stick deflection, lateral, deg

δ_{c_p} automatic-pilot control stick deflection, fore and aft, deg

δ_e elevator deflection, deg

δ_r rudder deflection, deg

δ_s servo drum rotation, deg

θ angle of pitch, deg

σ_θ pitch tracking error, mils, positive when target is above the tracking line

σ_ψ yaw tracking error, mils, positive when target is to the right of the tracking line

ϕ angle of bank, deg

ψ angle of yaw, deg

ω circular frequency, radian/sec

Subscripts:

e elevator

a ailerons

r rudder

A dot placed over a symbol indicates differentiation with respect to time.

DESCRIPTION OF AIRPLANE AND AUTOMATIC PILOT

Airplane

The airplane used was a Grumman F9F-2 (BuAero. no. 122560). This airplane has a straight wing, is powered by a turbojet engine, and is of conventional configuration. A photograph of the airplane is presented in figure 1 and a two-view drawing of the airplane is shown in figure 2. General dimensions and characteristics of the airplane are listed in table I. The wing-tip fuel tanks were on the airplane for all flights but no fuel was carried in them. A hydraulic booster system, which provides a boost ratio of approximately 37:1, is incorporated in the aileron control system of the airplane and a spring tab is used in the elevator control system. The rudder control system is of the conventional manual type.

Some data on the response characteristics of the airplane alone are presented in frequency-response form in figure 3. Figure 3(a) presents longitudinal frequency-response data in terms of θ/δ_e and figure 3(b) shows lateral frequency-response data in terms of ϕ/δ_{a_T}. As indicated on the figures, the data are for two different flight conditions. The frequency-response curves are quite normal for this type of airplane and, except for low damping of the Dutch roll oscillation and rather high longitudinal control forces, the flying qualities of the airplane were good.

Automatic Pilot

The automatic pilot used was basically a General Electric G-3 model. This automatic pilot is all electric in operation and of the attitude type. Except for the servo motors which operate on direct current the automatic pilot operates on alternating current. A quite detailed description of the components and of the operation of a standard model of this automatic pilot is given in reference 3. The automatic pilot used in the flight program reported herein differed in certain details from a standard G-3 model. The major changes were: the standard G-3 automatic-pilot controller was replaced by a control stick which simulated a conventional manual type of control stick both as to location and motion; the method of introducing signals into the servo amplifier from the automatic-pilot controller was changed (with the standard G-3 automatic pilot the signals from the automatic-pilot controller reach the servo amplifier with time lag and with the modified system the controller signals are introduced directly into the servo amplifiers); and rate gyros were added to the pitch and roll channels of the automatic pilot.

Block diagrams of pitch, roll, and yaw channels.- Block diagrams of the pitch, roll, and yaw channels of the automatic pilot in the maneuvering mode of operation are shown in figure 4. Figure 4(a) shows a block diagram of the pitch channel and figure 4(b) shows block diagrams of the roll and yaw channels.

In pitch, for steady-state conditions, the airplane pitch angle as measured by the vertical gyro is proportional to the fore or aft position of the automatic-pilot stick. The rate gyro and servo follow-up and tachometer signals provide stability and damping to the system. The servo follow-up canceler is a positional servomechanism having a relatively long time constant. For steady-state conditions the output of the servo follow-up canceler is equal in magnitude and opposite in sign to the servo follow-up signal. Steady-state servo follow-up signals such as result from changes in the elevator deflection required for balance with change in flight condition (airspeed, altitude, center-of-gravity location, etc.) are thus effectively canceled and the steady-state pitch-attitude angle is therefore independent of elevator position. Since the servo follow-up canceler has a relatively long time constant, it has little influence for rapid motions. In the pitch channel the response and damping characteristics of the airplane-automatic-pilot combination can be varied by changing the gains of the rate gyro and servo follow-up signals and the automatic-pilot stick sensitivities.

No independent adjustment of the servo tachometer signal gradient is provided but rather a constant ratio of servo follow-up gain to servo tachometer gain is maintained.

The operation of the roll channel (lower part of fig. 4(b)) is substantially the same as that of the pitch channel. The differences are that no servo feedback canceler is used and an additional signal source is present. The additional signal comes from a directional gyro, which provides heading stabilization. The directional gyro signal is cut out when the automatic-pilot control stick is moved laterally. In the roll channel the servo does not actuate the ailerons directly but rather actuates the input of the hydraulic boost unit in the aileron control system. The same signal gradients or gains which were adjustable in the pitch channel, previously described, are also adjustable in the roll channel.

A block diagram of the yaw channel is shown on the upper part of figure 4(b). The human pilot does not introduce signals into this channel of the automatic pilot. The yaw channel receives its operating signals from a rate gyro which increases the damping in yaw of the airplane and a pendulum, the purpose of which is to regulate to zero the lateral acceleration acting on the airplane. The operation of the canceler system in the yaw channel is substantially the same as that in the pitch channel which was described previously. As can be seen from the block diagram, the yaw-rate gyro signal is introduced into the canceler system as well as directly into the servo amplifier. The rate gyro signal is thus effectively canceled when the airplane is in a steady turn. In addition, the canceler reduces any steady-state rudder servo follow-up signals to zero. Again the canceler system has a relatively long time constant and therefore has little effect for rapid airplane motions. In the yaw channel the rate gyro, pendulum, rudder servo follow-up, and canceler tachometer gains are adjustable.

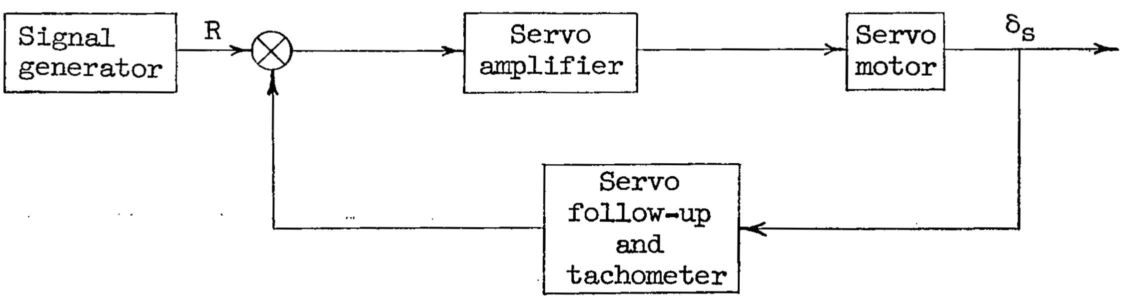
Automatic-pilot controller.- The human pilot introduced signals into the automatic pilot by moving a control stick, the grip of which was located in about the same position as that of the airplane conventional control stick. Longitudinal or lateral stick motions generated electrical signals proportional to the stick deflection and these signals were introduced directly into the pitch or roll servo amplifiers. Figure 5 shows a photograph of the control-stick installation. When the automatic-pilot control system was being used, the upper part of the conventional control stick was removed so as to avoid interference between the two sticks. The stub and lower part of the conventional control stick remained connected to the control surfaces of the airplane when the automatic pilot was being used. The automatic-pilot control stick was shorter than a conventional control stick, being about 15 inches long. The maximum stick throws were about $\pm 20^{\circ}$ in a longitudinal direction and about $\pm 18^{\circ}$ in a lateral direction. The stick sensitivities (ratio of electrical signal output to stick deflection) could be varied.

However, when the stick sensitivity was varied the maximum pitch and bank angles obtained for full stick deflection would be changed proportionally. Therefore, the stick sensitivity could not be reduced without reducing the maximum pitch and bank angles attainable. The flight test results presented in this paper were obtained with constant stick sensitivities. For the stick sensitivities used, full lateral stick deflection produced a steady-state bank angle of about 60° and full longitudinal stick deflection produced a pitch angle of 35° to 40°. There was no mechanical connection between the automatic-pilot control stick and the airplane control system; therefore motions of the airplane control surfaces were not transmitted to the stick.

Two types of stick-force feel were used with the attitude control system for both fore-and-aft and lateral stick motions. One of the feel systems provided a force to the pilot which was proportional to stick deflection (spring feel) and the other provided a force proportional to the rate of stick deflection (damping feel). Several spring rates were used with the spring feel system. Figure 6(a) shows the variation of longitudinal stick force with stick position for one of the spring rates used, and figure 6(b) shows a similar plot for lateral stick motions. About one pound of preload was used to overcome the friction and thus provide stick centering.

The characteristics of the damper feel system used are shown in figures 7(a) and 7(b) by the variation of longitudinal and lateral stick force with rate of stick deflection. About one-half pound of friction was present with both the longitudinal and lateral damper feel systems. For longitudinal stick motions the push forces required to produce a given rate of stick motion were lighter than the pull forces. This characteristic was unintentional.

Automatic-pilot response characteristics.- Ground tests were made of some of the automatic-pilot components in order to determine their response and damping characteristics for various operating conditions. The results of some frequency-response tests of the automatic-pilot servo loop are presented in figure 8. A block diagram of the system used in determining the servo loop frequency response is shown below:



Tests were made with the servo motor operating under no load and with a spring and inertia load, with various gains on the servo follow-up signal, and with various magnitudes of input signal. The electrical input signal was generated by a synchro transmitter which was driven by an electric motor through a variable-speed reduction drive. In the ground tests made with the servo loaded, the moment of inertia of the longitudinal control system of the F9F-2 airplane about the elevator hinge line was approximated, and a spring which provided a servo torque-deflection gradient about the same as that present for the elevator servo in the test airplane when flying at a Mach number of 0.7 and an altitude of 30,000 feet was used.

The amplitude ratio data of figure 8 are presented in terms of servo drum rotation in degrees to input signal in volts. The input voltage can be converted to automatic-pilot stick motions by using the following constants: 0.8 volt per degree of lateral stick motion and 0.55 volt per degree of longitudinal stick motion. Assuming the airplane control systems to be represented by simple gains and neglecting control system stretch, servo drum rotations can be converted to control-surface motions by use of the following constants:

Elevator - 0.2 degree δ_e per degree δ_s

Total aileron - 0.6 degree δ_{a_T} per degree δ_s

Rudder - 0.23 degree δ_r per degree δ_s

Figure 8(a) presents data obtained using three amplitudes of input signal with the servo operating under load. The values of inertia and spring load used are listed on the figure. The value of servo follow-up gain used (3.5 volts per radian of servo drum rotation) is about the same as that found to be satisfactory (in combination with other automatic pilot settings) for the elevator servo for high-altitude flight (h_p = 30,000 ft). The data of figure 8(a) show the servo loop to be well damped and to have a natural frequency of about 2 cycles per second. Also, the differences in the amplitude-ratio and phase-angle curves show the servo to be somewhat nonlinear in operation. At frequencies below approximately 2 cycles per second, the amplitude ratio increases as the magnitude of the input signal increases and at frequencies greater than about 2 cycles per second the opposite occurs.

Figure 8(b) shows a comparison of the frequency response of the servo loop when it is operating with no load and with the combination spring and inertia load. The servo follow-up gain was the same as that used for the data presented in figure 8(a), and the same magnitude of input signal was used for both the servo loaded and unloaded conditions. Inspection of figure 8(b) shows that when the servo was operating under load the amplitude ratio at low frequency was reduced about 25 to 30 percent from that obtained with no load on the servo. This reduction in amplitude ratio with load occurred for all magnitudes of input signal used. The range of input signal used was about 0.5 to 1.5 volts. Ground tests were made using only one value of inertia and spring load and therefore the overall variation of amplitude ratio with load is not known. There were no significant differences in the phase angles between the servo loaded and unloaded conditions.

Figure 8(c) shows the effect of varying the servo follow-up gain on the frequency-response characteristics. The amplitude of the input signal used was approximately the same with either gain and also the servo was operated under load for these tests. As expected, the effect of increasing the servo feedback gain is to reduce the amplitude ratio at low frequency and to increase the natural frequency. As was previously mentioned, the lower feedback gain (3.5 volts per radian δ_s) is about the same as that used for the elevator servo in high-altitude flight ($h_p = 30{,}000$ ft). The higher feedback gain (6.7 volts per radian δ_s) is near the maximum available and is approximately the same as that used for the elevator servo for low-altitude flying, and for the aileron and rudder servos for all flight conditions. For either value of follow-up gain the ratio of the maximum amplitude ratio to the static sensitivity has about the same value of 1.5. This constant ratio results because a constant ratio of servo tachometer gain to servo follow-up gain is maintained when the follow-up gain is varied.

All the automatic-pilot servos were located in the fuselage near the cockpit at considerable distances from the control surfaces. There was therefore considerable stretch in the rudder and elevator control systems which are of the cable type. The effect of the control system stretch would be to reduce the gain of the automatic-pilot control system. The greatest stretch occurred in the rudder control system and when flying at a Mach number of 0.6 at an altitude of 10,000 feet the rudder deflections were about 0.6 what they would have been if no stretch had been present. The aileron control system is of the push-rod type and is considerably stiffer than the rudder and elevator systems.

Data on the speed-torque characteristics of the automatic-pilot servo motor are presented in figure 9. With no load the servo drum rotational speed is 360° per second. The servo stall torque is 160 to 180 inch-pounds.

Ground tests were also made to obtain the transient response characteristics of the canceler system in the pitch channel of the automatic pilot. The tests consisted of applying near step voltage inputs to the canceler system amplifier and measuring the output voltage of the canceler synchro transmitter. Figure 10 shows time histories of the input to the canceler system and the output of the canceler system for two magnitudes of input signal. Inspection of figure 10 shows the response characteristics of the canceler system to be nonlinear with a time delay and speed limiting being present. The time constants of the canceler systems in both the pitch and yaw channels of the automatic pilot could be varied. The data shown in figure 10 were obtained with the same setting as was used in flight for the pitch channel. The time constant of the yaw canceler system was about 2 to 3 times larger than that of the pitch canceler system.

Some data on other automatic-pilot components are listed below:

Automatic-pilot component	Natural frequency	Damping ratio	Range	Signal gradient
Vertical gyro: Pitch	------	----------	±60°	0.27 volt/deg
Roll	------	----------	±60°	0.25 volt/deg
Directional gyro	------	----------	-------------	0.4 volt/deg
Rate gyros	20 cps	0.6	±1 radian/sec	Variable (maximum = 0.4 volt/deg/sec)
Pendulum	------	0.4 to 0.6	±0.07 g	Variable (maximum = ±18.9 volts/g)

The pendulum was located about 5 feet forward of the center of gravity of the airplane in the nose-wheel well of the airplane.

INSTRUMENTATION

NACA recording instruments, which measured the following quantities, were installed in the airplane:

Normal, longitudinal, and transverse accelerations
Pitching, rolling, and yawing velocities and accelerations
Airspeed and altitude
Elevator, aileron, and rudder positions
Elevator, aileron, and rudder servo positions
Angle of attack and sideslip angle
Pitch and bank attitude angles
Longitudinal and lateral automatic-pilot control stick positions
Longitudinal and lateral automatic-pilot stick forces

The airspeed head, which was used to measure airspeed and altitude, was mounted on a boom which extended out of the nose of the airplane. (See fig. 1.) No calibration was made of the airspeed installation and therefore the airspeed and altitude data presented in this paper have not been corrected for position error. It is estimated that the error in the measured static pressure due to the fuselage pressure field is about 2 percent of the impact pressure at low angles of attack. The airplane angle of attack and sideslip angle were measured with vanes which also were mounted on the nose boom.

For tracking flights a 16-millimeter camera was used to photograph the gunsight image and a reflected image of the target airplane in order to obtain a record of the tracking errors.

FLIGHT TESTS, RESULTS, AND DISCUSSION

The characteristics of the airplane—automatic-pilot system were evaluated in flight by making various maneuvers such as abrupt and constant acceleration pull-ups, abrupt rolls, turns, and rudder kicks. Data were also obtained during various flight operations such as air-to-air tracking, ground strafing runs, rough-air flying and landings. In order to have a basis for comparison, many of the flight operations were also performed when the airplane was controlled through the conventional system.

Characteristics in Pitch

Transient response characteristics.- The response characteristics of the airplane—automatic-pilot system in pitch were determined for various flight conditions by abruptly deflecting the automatic-pilot control stick and recording the airplane response. Figure 11 shows time histories of automatic-pilot stick position and stick force, elevator position, pitch attitude angle, and normal acceleration in maneuvers performed at various flight conditions as noted on the figure. Some changing of the pitch-rate gyro and servo feedback gains was found to be necessary with change in flight condition. The values of the gains used are listed on the figure. At an altitude of 30,000 feet, figures 11(a) and 11(b), the same values of servo follow-up and pitch-rate gyro gains were used at Mach numbers of 0.60 and 0.76. At an altitude of 5,000 feet, figures 11(c), (d), and (e), the same servo follow-up gains were used throughout the speed range but the pitch-rate gyro gain was reduced for the highest speed. The gains used are not necessarily optimum but they were considered by the pilot to be satisfactory from the standpoint of response and damping. All the maneuvers shown were made with the damper feel system installed. Similar maneuvers have been made when using the spring feel system. For a given stick movement the airplane response would, of course, be the same when using either the damper or spring feel systems. Also, if the spring feel system had been used the stick-force curves would be substantially the same as the stick-position curves.

The response and damping as shown by the pitch attitude angle and normal acceleration time histories of figure 11 are in general satisfactory for the range of flight conditions investigated. For Mach numbers of about 0.6 or greater at both low and high altitude the response times (time for pitch-attitude angle to reach and stay within 10 percent of the commanded steady-state value) are on the order of 1.5 to 2.0 seconds with the shortest response time occurring at the highest dynamic pressure. In the power approach condition at an indicated airspeed of 125 knots, figure 11(c), the damping is lower than for the other flight conditions as is indicated by the somewhat oscillatory nature of the response. All the pilots who flew with the attitude control system preferred a response in which there was little or no overshoot of the commanded steady-state attitude angle.

With a conventional control system the fore or aft stick motions are, of course, substantially the same as the elevator motions. A comparison of the automatic-pilot stick-position curves in figure 11 with the elevator-position curves shows that the stick motions required to produce a change in attitude angle as shown in the figure are considerably different and simpler with an attitude control system than with a conventional control. Pilots adapted themselves to the attitude control system quite easily. However, the pilots did not consider the simpler stick motions used with the attitude control system to offer any significant advantage. Furthermore, the fact that the automatic-pilot control stick did not follow the control-surface motions was not objectionable to the pilots. The lower part of the conventional control stick, which was connected to the control surfaces, was visible to the pilots but they did not consider it of any advantage to watch the motions of this stick. However, with some systems it may be desirable to provide indications of control-surface positions to the pilot.

Frequency response.- Frequency analyses were made of transient responses, such as presented in figure 11, in order to obtain frequency-response data. The frequency analyses were made using a Coradi harmonic analyzer. For a description of this machine and the analyses procedure, see reference 4. Automatic-pilot stick position and stick force were used for input quantities and pitch attitude angle and normal acceleration were used for output quantities. Figure 12 presents frequency-response data for a Mach number of 0.60 and an altitude of 30,000 feet. The damper feel system was used in the maneuver for which data are presented in figure 12; however, the θ/F_{c_p} and a_n/F_{c_p} amplitude ratio curves for the spring feel system would be expected to have approximately the same shape as the θ/δ_{c_p} or a_n/δ_{c_p} curves. The amplitude ratios θ/δ_{c_p} and a_n/δ_{c_p} can be converted to θ/F_{c_p} and a_n/F_{c_p} by dividing by the spring gradient (stick force per unit stick deflection) of the feel system.

Pitch-attitude-angle response: As can be seen from figure 12(a), and as has been discussed previously, with the attitude control system, a static sensitivity exists between pitch-attitude angle and automatic-pilot stick position. With a conventional control system, assuming constant speed, a static sensitivity exists between pitching velocity and elevator (or stick) position. At high frequency with a conventional control system or with an attitude control system, if a perfect servo is assumed, the airplane pitching angular acceleration is approximately in phase with the elevator (or stick) motion. The phase angles at high frequency between pitching velocity and stick position and pitch angle and stick position are therefore -90° and -180°, respectively. In figure 13(a) the phase angles between θ and δ_{c_p} are greater than -180° at high frequency. The phase angles greater than -180° can be attributed to the servo, canceler system, etc.

The frequency-response data θ/F_{c_p} for the attitude control having the damper feel system are presented in figure 12(b). With the damper feel system the stick force approaches zero as the frequency approaches zero and the amplitude ratio θ/F_{c_p} therefore approaches infinity as the frequency approaches zero. Also since the stick force is in phase with the rate of stick motion the phase angles between θ and F_{c_p} are approximately 90° greater than between θ and δ_{c_p} throughout the frequency range.

Normal acceleration response: The frequency-response data a_n/δ_{c_p} $\left(a_n/F_{c_p}\right.$ for the spring feel system$\left.\right)$ and a_n/F_{c_p} for the damper feel system are presented in figures 12(c) and 12(d). With an attitude control system in order to make a constant acceleration pull-up, which corresponds to zero frequency on a frequency-response basis, the pilot must move the automatic-pilot control stick back at a constant rate. The stick deflection (and stick force for a spring feel system) therefore increases with time during the pull-up. As can be seen in figure 12(c) this causes the amplitude ratios a_n/δ_{c_p} or a_n/F_{c_p} (for a spring feel system) to approach zero at zero frequency. The inverse of a_n/F_{c_p} or force per g therefore approaches infinity in steady pull-ups and rapidly decreases as the frequency increases. From a flying qualities standpoint this means that the force per g in steady pull-ups is greater than in rapid pull-ups. Past research (see, for example, ref. 5) has indicated that this is an undesirable characteristic and the Air Force-Navy Flying Qualities Specifications (ref. 6) require that the force per g in rapid pull-ups should not be less than in steady pull-ups. With the damper feel system, figure 12(d), the amplitude ratio a_n/F_{c_p} is a maximum at zero frequency so that the force per g, that is, F_{c_p}/a_n has a minimum value in steady pull-ups and becomes larger for rapid pull-ups. Only one value of damping was used with the damper feel system. In steady pull-ups a stick force of about one pound per g was supplied by the damper. In the pilot's opinion this value of force per g was somewhat light. However, it was the pilot's opinion that the steady force per g should probably be lighter with the attitude control system having damper feel than with a conventional control system. One reason for this is that the stick forces required in rapid pull-ups are considerably higher than in steady pull-ups whereas with most conventional control systems this is not the case. With a conventional control system the phase angle between normal acceleration and elevator deflection is 0° at zero frequency and 180° at high frequency. With an attitude control system, assuming a perfect servo, the phase angle between normal acceleration and automatic-pilot stick position approaches 90° at zero frequency and -180° at high frequency. The phase angle between normal acceleration and stick force for the damper feel system is 0° at zero frequency and and -270° at high frequency.

It is hoped that, by accumulation of data of the type presented in figure 12 and comparison of the data for various systems, a more rational specification for the dynamic characteristics of feel forces can be established. Not much can be concluded as yet, since this is the first attempt to analyze the data in this way.

<u>Control forces in maneuvers with spring and damper feel systems</u>.- This section of the paper describes the differences in control forces required in maneuvering when the spring feel system and the damper feel system are used. Figure 13 shows two similar pitching maneuvers, one made when using the spring feel system, and the other when using the damper feel system. If the pilot makes a pull-up when using the spring feel system and then reduces his pull force, as was done at about time 6 seconds in the maneuver shown in figure 13(a), the airplane may very likely develop a negative acceleration. The pilot is not required to apply a push force to produce the negative acceleration and therefore he can very easily inadvertently induce it. In the particular maneuver shown, only a small value of negative acceleration was reached but had the pilot reduced his pull force more rapidly an appreciable negative acceleration would have occurred. This characteristic of the spring feel system was very objectionable to the pilot.

When the damper feel system was used, this undesirable characteristic was eliminated. As can be seen from figure 13(b), in the maneuver with the damper feel system, when the pilot reduces his pull force to zero (stops the aft stick motion), the airplane returns to 1 g flight. Furthermore the pilot must apply a push force to produce an acceleration less than 1 g and therefore he is not likely to induce a negative acceleration inadvertently.

In constant-acceleration turns with the damper feel system, the pilot is not required to apply a force to the stick once the acceleration is established and thus has zero force per g. With the automatic-pilot control system used the maximum acceleration obtainable in steady turns is 2g ($\phi = 60^{\circ}$). In turns at this relatively low level of acceleration the pilots had no objection to the lack of a force per g. Whether the lack of a force per g would be objectionable in steady turns at higher levels of acceleration is not known. With the spring feel system, in constant acceleration turns, a pull force is required since the control stick must be moved back to provide an electrical signal to balance the signal generated by the pitch-rate gyro. The pull force increases with increase in acceleration since the pitching angular velocity and thus the pitch-rate gyro signal increases with increase in acceleration. Also, since the pitching velocity per unit of acceleration is inversely proportional to true airspeed, the stick force per unit of acceleration decreases with increase in airspeed. In constant rate-of-climb turns an additional increment of aft stick deflection (and therefore an additional pull force) is required to maintain the climb angle. In diving turns an increment of push force is required to maintain the dive angle.

The advantages of the damper feel system over the spring feel system in providing higher forces in rapid pull-ups than in steady pull-ups have been discussed earlier in the paper.

Figures 13(a) and (b) illustrate another difference in the flight characteristics provided by the spring and damper feel systems. With the spring feel system, figure 13(a), if the pilot makes a pull-up and changes the airplane attitude angle as was done in the first part of the maneuver, the pilot must apply a force to maintain the new attitude angle. This characteristic was objectionable to the pilots when they were required to hold the force for long periods of time. The pilots' objections could probably be overcome by providing a means of trimming out the pilots' force at a slow rate. With the damper feel system, figure 13(b), the pilot must apply a force only when moving the control stick and for any steady attitude angle no stick force is required.

Another characteristic of the attitude control having the damper feel system which is different than with a spring feel system or a conventional control is that, if when in trimmed steady level flight the pilot moves the automatic-pilot control stick fore or aft from neutral, the stick will not return to its original trim position. The airplane will therefore have no tendency to maintain the original trim speed but rather will dive or climb and the airspeed will increase or decrease. In the opinion of the pilot who did most of the flying with the system tested, the lack of speed stability was not an important factor. However, this opinion is based on limited experience with the system and for some flight operations the lack of speed stability may be objectionable. If the speed stability is found to be a desirable or necessary characteristic, it can probably be provided by installing a bungee or a spring with a small spring rate in parallel with the damper feel system. Then, if the pilot is applying no force to the stick, the stick will return toward neutral at a slow rate.

Characteristics in Roll

Transient response characteristics.- The response characteristics of the airplane—automatic-pilot combination in roll for abrupt lateral stick deflections are presented in figure 14. All the maneuvers shown in figure 14 were made when using the damper feel system. The time histories of the maneuvers shown in figure 14(a) were obtained at a Mach number of 0.6 and an altitude of 30,000 feet, and are for three magnitudes of stick deflection. Inspection of figure 14(a) shows that for the two smaller amplitudes of stick deflection the response and damping are good. For the largest amplitude input ($\phi = 40^\circ$ left to 50° right) there is approximately 15° overshoot of the bank angle. The pilots objected to the overshoot and preferred a response where little or no overshoot occurred. As was mentioned previously, the range of the rate gyros used was ±1.0 radian per second and therefore for the largest amplitude input where a rolling velocity of about 2.3 radians per second was reached, the relative damping supplied by the rate gyro was reduced. Also, there is a possibility that servo rate limiting occurred in this maneuver.

Figures 14(b) and (c) show the response characteristics in roll for other flight conditions as noted on the figures. Again, some gain changing of the roll rate gyro was found necessary for the various flight conditions. The gains used are listed on the figures. The same roll rate gyro gain was used at Mach numbers of 0.6 and 0.7 at an altitude of 30,000 feet (figs. 14(a) and (b)). The roll rate gyro gain at $V_i = 125$ knots and at $M = 0.6$ at an altitude of 5,000 feet (figs. 14(c) and (d)) was also the same. At the higher speeds ($M = 0.70$, $h_p = 30{,}000$ ft and $M = 0.6$, $h_p = 5{,}000$ ft), figures 14(b) and (d), there is some lateral unsteadiness present as shown by the high-frequency small-amplitude oscillations of the ailerons.

The pilots had some objections to the type of roll response provided by the attitude control system. The main objection was that the response seemed jerky for small, rapid, or irregular stick motions. One basic reason for the feeling of jerkiness may be that, with an attitude control system, the human pilot does not control the airplane control surfaces directly as with a conventional control and he therefore has less direct control over the airplane angular accelerations and angular velocities. With the particular system used, small rapid stick motions produced larger rolling accelerations than did the conventional control for similar pilot inputs and this contributed to the feeling of jerkiness. Also, larger rolling accelerations than ordinarily used by the pilot with conventional controls are present in stopping the rolling motion near the commanded steady-state bank angle. Increasing the damper feel forces on the stick alleviated the feeling of jerkiness somewhat, since the pilot then tended to move the stick more smoothly. However, increasing the damper feel forces made it more difficult for the pilot to make large rapid bank angle changes because of the higher stick forces required. This difficulty might be eliminated by use of a nonlinear damper.

Several methods can be used to overcome the feeling of jerkiness. In general, they operate on the principle of limiting the rate at which the input signal from the human pilot is sent to the servo motor. The result is that the response of the airplane is slowed down and in maneuvering the control system is similar to a slow rate control rather than an attitude control. If rapid maneuvering through large attitude angles is required, it may not be desirable to restrict the maneuvering rates.

The question arises as to why the feeling of jerkiness or oversensitivity which the pilots objected to in rolling maneuvers was not as noticeable in pitching maneuvers. At least a part of the sensitivity problem in roll might result because the electrical signal output per degree of stick deflection was larger for the roll channel than for the pitch channel. It was not practicable to reduce the ratio of electrical signal output to stick deflection in the roll channel because this would have reduced the already limited maximum bank angles obtainable. Also, because of the hydraulic booster in the aileron control system, higher rates of aileron motion than elevator motion could be obtained. Furthermore, from airplane geometric and mass considerations the ailerons are inherently capable of producing larger rolling accelerations than the elevator is of producing pitching accelerations and the rolling accelerations are produced more rapidly than normal accelerations.

Frequency response.- Frequency-response data, similar to that previously presented for the airplane—automatic-pilot system in pitch, were also obtained in roll. Figure 15 presents frequency-response curves of ϕ/δ_{c_l} and ϕ/F_{c_l} for a Mach number of 0.60 and an altitude of 30,000 feet. These data were obtained from a maneuver in which the damper feel system was used. The comments made earlier in the paper concerning the frequency-response characteristics in pitch are, in general, applicable also to the roll-frequency-response data.

As was the case in pitch, the pilots much preferred the damper feel system to the spring feel system for lateral control stick motions. One objection to the spring feel system is that the pilot is required to apply a force to maintain any constant bank angle other than zero. This characteristic was particularly objectionable when the stick force was large. With the damper feel system or with a conventional control, no control force is required for a steady bank angle.

Dynamic Lateral Stability

Time histories of the short-period lateral oscillation for the airplane alone and for the airplane with the yaw channel of the autopilot operative are shown in figure 16. The oscillations were induced by the pilot by deflecting the rudder pedal and then releasing it. When the yaw channel of the autopilot was operative, the pilot overpowered the servo when deflecting the rudder. The maneuvers shown in figure 16 were made at a Mach number of 0.60 and an altitude of 30,000 feet. A comparison of the two maneuvers shows the yaw channel of the automatic pilot to be very effective in increasing the damping of the lateral oscillation. Also, no measurable residual oscillations resulted from use of the yaw channel.

Rough-Air Flying, Tracking, and Landing Characteristics

Rough-air characteristics.- Figures 17(a) and 17(b) are time histories of portions of two runs, one with the automatic control system and one with the conventional control system, made when flying in rough air at a Mach number of 0.6 at an altitude of 5,000 feet. For the run made with the conventional control system, the pilot maintained straight and level flight with the precision ordinarily used for cross-country flying. For the run with the automatic control system no command inputs were used.

As has already been shown, the Dutch roll motion of the airplane is lightly damped at high altitude. Although the damping is greater at an altitude of 5,000 feet than at 30,000 feet, the damping is still low enough that the Dutch roll oscillation is objectionable in rough air. In order to improve the handling qualitites of the airplane, the yaw channel of the automatic pilot was used for the run with the conventional control system presented in figure 17(b).

An examination of figure 17(b) shows that with the human pilot controlling the airplane, it responded to gusts primarily in bank angle and normal acceleration. When on automatic control (fig. 17(a)), the automatic pilot regulated the bank angle much better than the human pilot did with the conventional control system but the autopilot had little effect on the normal acceleration. The pitching and yawing motions were quite small in either case.

The effect of the automatic pilot on the motion of the airplane can be explained by consideration of the quantities sensed by the elevator and aileron channels. The aileron channel which sensed the bank angle, rolling velocity, and heading, could effectively regulate the bank angle and provide long-period heading stabilization. The elevator channel sensed the pitch angle and pitching velocity but not normal acceleration or angle of attack. Since the pitching velocity and pitch angle were quite steady even without the automatic pilot, the automatic pilot had little effect on the longitudinal motion of the airplane.

It should be noted that the ratio of directional gyro-signal gradient to roll vertical gyro-signal gradient of this automatic pilot was small, being about 1.6. Higher values of this ratio, such as might be used in fully automatic interceptors, would give larger variation of the bank angle.

For flight in rough air the pilot greatly preferred the automatic control system to the conventional control system. The attitude stabilization of the automatic pilot relieved the pilot of the necessity of making control corrections almost continuously and, in addition, maintained the bank and heading attitudes better than he could with the conventional control system.

Tracking.- Tracking runs on a target airplane and ground strafing runs in rough air were made to evaluate quantitatively the automatic pilot-control system when the pilot was performing precision tasks. For comparison purposes, similar runs were made with the pilot controling the airplane with the conventional control system. For all tracking runs made with the conventional control system, both air-to-air and strafing, the yaw channel of the automatic pilot was in operation.

A fixed optical gunsight was used in the tracking tests and a 16-millimeter gun camera was used to photograph the gunsight presentation. The gunsight camera records were evaluated in terms of the standard deviations of the pitch and yaw sighting errors.

Air-to-air tracking: The following maneuvers were used for the air-to-air tracking: nonmaneuvering tail chase, 30° to 50° banked turns, pull-ups to $2\frac{1}{2}$g and push-downs to about $\frac{1}{4}$g. These are relatively mild maneuvers such as might be used by a bomber-type airplane. The duration of the maneuvers was about 30 to 45 seconds for the nonmaneuvering tail chase and turns, and about 10 seconds for the pull-ups and push-downs. All the air-to-air tracking runs were made at a Mach number of about 0.6, an altitude of about 30,000 feet, and a range of about 500 yards. The gunsight (and therefore the aiming line established by the gunsight) was elevated 2° from the fuselage reference line for the air-to-air tracking. This was done for two reasons: First, it placed the tracking airplane below and well out of the wake of the target airplane; and second, it allowed the pilot to move the tracking line laterally by rolling the airplane. Thus, when corrections were made for yaw errors, the tracking line led the yaw angle of the airplane.

Table II shows a comparison of the tracking errors for various maneuvers when using the conventional control and when using the automatic-pilot control having the damper feel system. The tail chases and turns represent slightly over one minute of tracking time for each case and the pull-ups and push-downs about 1/2 minute.

In general, there are no significant differences in the pilot's tracking ability with the two systems, the pitch errors being slightly larger and the yaw errors being slightly smaller for the automatic control system. It should be pointed out that the pilot had little inducement to reduce the tracking error to less than about 1 to $1\frac{1}{2}$ mils. For example, the tailpipe diameter of the target airplane appeared to be of about this size on the gunsight at the tracking range used.

Figure 18 shows time histories of two tracking runs in turns, one made when using the conventional control system and the other when using the automatic-pilot control system. Examination of the pitch tracking error and the normal-acceleration time histories in figure 18 reveals an irregular oscillation with a period of $1\frac{1}{2}$ to 2 seconds to be present with both of the control systems. The oscillation is more noticeable for the run with the automatic-pilot control system. The force per unit acceleration with the automatic-pilot control was about 1 pound per g, which in the pilot's opinion was rather light. The force per unit acceleration with the conventional control was about 9 to 10 pounds per g. The light stick forces present with the attitude control may have contributed to the larger oscillations present with this system. Examination of the bank-angle time histories in figure 18 shows the bank-angle time histories to be smoother with the automatic-pilot control.

In addition to the air-to-air tracking with the damper feel system, several tracking flights were made with the spring feel system. Although several spring gradients were tried in both pitch and roll, no system was found that the pilot considered satisfactory. With the exception of nonmaneuvering tail chases, tracking errors with the spring feel system were two or three times larger than those for the damper feel system or the conventional control system.

Strafing: Strafing runs on a fixed ground target were used to evaluate the automatic-pilot control system in rough air. For comparison purposes, similar runs were made when using the conventional control system with the rudder channel in operation. All runs were made at a Mach number of about 0.6 at altitudes from 3,000 feet down to about 600 feet. For the strafing tests the tracking line was parallel to the fuselage reference line. The turbulence was termed heavy to occasionally moderate by the pilot. The tracking errors in table II represent slightly over a minute of time on target for each system.

Two typical strafing runs are presented in figure 19. It can be noted that the variations in bank angle for the run with the automatic pilot (fig. 19(a)) are smaller than those for the run with the conventional control system (fig. 19(b)). Otherwise, the automatic pilot had little effect on the motions of the airplane.

As in the air-to-air tracking there was no appreciable difference in the pilot's tracking ability with the two systems. It may be thought that the attitude stabilization of the automatic pilot would make the airplane a more stable gun platform and hence improve the tracking. That it did not was probably due to the fact that there was not much displacement of the airplane except in bank, and displacement in bank does not necessarily introduce sighting errors. Although the sighting errors in rough-air strafing runs were about the same with the attitude or conventional control systems, the pilot preferred the attitude control system for this flight operation and, in fact, thought that he could do a better job with this system. The airplane was steadier, particularly in bank, and therefore the pilot was not required to make bank-angle corrections almost continuously.

One feature of both control systems which the pilots found undesirable, both in strafing and in air-to-air tracking, was their inability to make small corrections in yaw by sideslipping the airplane. They were of the opinion that they could have done a better job of tracking if they had had some direct control of the rudder.

Landing.- A time history of a landing with the automatic control system is shown in figure 20(a). For comparison, a similar landing with the conventional control system is shown in figure 20(b). A power-on sinking type of approach was used for these landings. Touch down was at about 100 knots indicated airspeed. Despite the differences in control forces and control motions, no difficulty was experienced in making the landing with the automatic-control system.

One difference in piloting technique was noted. With the automatic control system the pilot did not pump the stick as is generally done with conventional control systems. Instead, the stick was moved in a series of small rearward steps which resulted in step-like changes in the pitch attitude of the airplane. This probably indicates that stick pumping occurs because it is the technique used in obtaining similar step-like changes in attitude angle with a conventional control system.

In the landings made with the automatic pilot the cross winds were small. Some form of rudder control would, no doubt, be necessary for landing in cross-winds of appreciable magnitude.

Pilots' Opinion of Attitude Control System as a Maneuvering Control

Although the pilots were able to perform the flight operations reported herein about as well with the attitude control having the damper feel system as with the conventional control, they, in general, did not like the attitude control system as well as the conventional control system for rapid maneuvering (such as required of a fighter airplane). For flying involving only mild maneuvering, the airplane attitude and heading stabilization provided by the automatic pilot greatly improved the flying qualities of the airplane. Also, for flying in rough air (either in cross-country flying or in strafing runs) the pilots much preferred the attitude control system to the conventional control because the required pilot effort was greatly reduced.

As has been indicated in preceding sections of this paper, the servos used were of rather low performance and the question arises as to whether the pilots were influenced adversely by the servo characteristics. Since the flight investigation reported herein was made, the same servos have been used in a rate automatic-pilot system and in an irreversible power-control system. In the pilots' opinion the flying qualities of the airplane with these systems were good; thus the pilots' objections to the attitude system cannot be attributed to the servos.

CONCLUSIONS

A flight investigation was made to obtain experimental information on the handling qualities of a fighter airplane which the human pilot controlled by supplying signals to an attitude type of automatic pilot. An automatic-pilot control stick which simulated a conventional control stick was used by the human pilot to introduce signals into the automatic pilot. The main conclusions reached as a result of this flight program are as follows:

(1) In general, the pilots did not consider the attitude control system to be as desirable for rapid maneuvering (such as required in air-to-air gunnery) as a conventional type of control system. For flight

operations which involve only mild maneuvers or practically no maneuvering, the airplane attitude and heading stabilization provided by the automatic pilot greatly improved the flying qualities of the airplane.

(2) In maneuvering with the attitude control system, the pilots much preferred the control-force characteristics provided by a damper feel system to those provided by a spring feel system.

(3) For the rough-air flying performed in the flight program (cross-country and strafing runs), the pilots much preferred the attitude control with the damper feel system to the conventional control. The main improvement was that the airplane was stabilized in heading and roll and the pilot was not required to make corrections almost continuously.

(4) For precision flying, such as tracking a nonmaneuvering or a mildly maneuvering target and in strafing runs, the pilot was able to do about equally well when using either the attitude control having the damper feel system or the conventional control system. When the spring feel system was used with the attitude control, the tracking errors were considerably larger.

(5) The pilots had some objections to the type of roll response provided by the attitude control system. The main objection was that the response was jerky for small, rapid, or irregular stick motions. The feeling of jerkiness may result from the magnitudes of the rolling accelerations resulting from small stick deflections being larger than usual and also the magnitudes of the rolling accelerations present in stopping the rolling motion at the steady-state bank angle being greater than normally used.

(6) Pilots were able to adapt themselves to the attitude control system easily and did not consider the difference in stick motions required in maneuvering with the attitude and conventional control systems to be of particular importance. For rapid stick motions, the pilots wanted an airplane response in which there was little or no overshoot of the commanded steady-state bank or pitch attitude angles.

Langley Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Langley Field, Va., January 9, 1956.

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5. Phillips, William H.: An Investigation of Additional Requirements for Satisfactory Elevator Control Characteristics. NACA TN 1060, 1946.

6. Anon.: Military Specification - Flying Qualities of Piloted Airplanes. MIL-F-8785(ASG), Sept. 1, 1954. (Amendment-1, Oct. 19, 1954.)

TABLE I

GENERAL AIRPLANE DATA

Wing:	
Span (with tip tanks), ft	37.99
Span (without tip tanks), ft	35.25
Area (without tip tanks), sq ft	250
Airfoil section	NACA 64_1-A012
Aspect ratio (without tip tanks)	4.97
Taper ratio	0.46
Incidence, deg	0
Dihedral, deg	4
Twist, deg	0
Sweep of 27-percent chord line, deg	0
Mean aerodynamic chord (M.A.C.), in.	89.45
Total aileron area, sq ft	18.44
Aileron travel, deg	19 up 14 down
Horizontal tail:	
Span, ft	17.21
Area (including elevator), sq ft	66.20
Elevator area, sq ft	19.20
Elevator travel, deg	18 up 15 down
Tail length, 25-percent M.A.C. of wing to elevator hinge line, ft	18.45
Vertical tail:	
Area (not including dorsal fin), sq ft	36.02
Rudder area, sq ft	8.54
Rudder travel, deg	±26
Miscellaneous:	
Length (excluding nose boom), ft	38.13
Weight, take-off (tip tanks empty), lb	14,460
Center-of-gravity position, take-off, percent M.A.C.	26.5
Center-of-gravity position, landing (1,000 lb fuel), percent M.A.C.	28.4
Engine	J42-P-8

TABLE II

STANDARD DEVIATIONS OF TRACKING ERRORS WITH ATTITUDE AUTOMATIC CONTROL SYSTEM AND CONVENTIONAL CONTROL SYSTEM

Maneuver	Pitch error, mils		Yaw error, mils	
	Automatic control system	Conventional control system	Automatic control system	Conventional control system
Nonmaneuvering tail chase	2.6	2.2	1.7	1.7
Turns, $\phi = 30^\circ$ to 50°	4.6	3.6	3.1	3.8
Pull-ups and push-downs, 2.5 to 0.25g	5.4	4.4	2.7	3.1
Strafing	5.1	4.0	7.3	6.9

Figure 1.- Grumman F9F-2 airplane.

L-71396.1

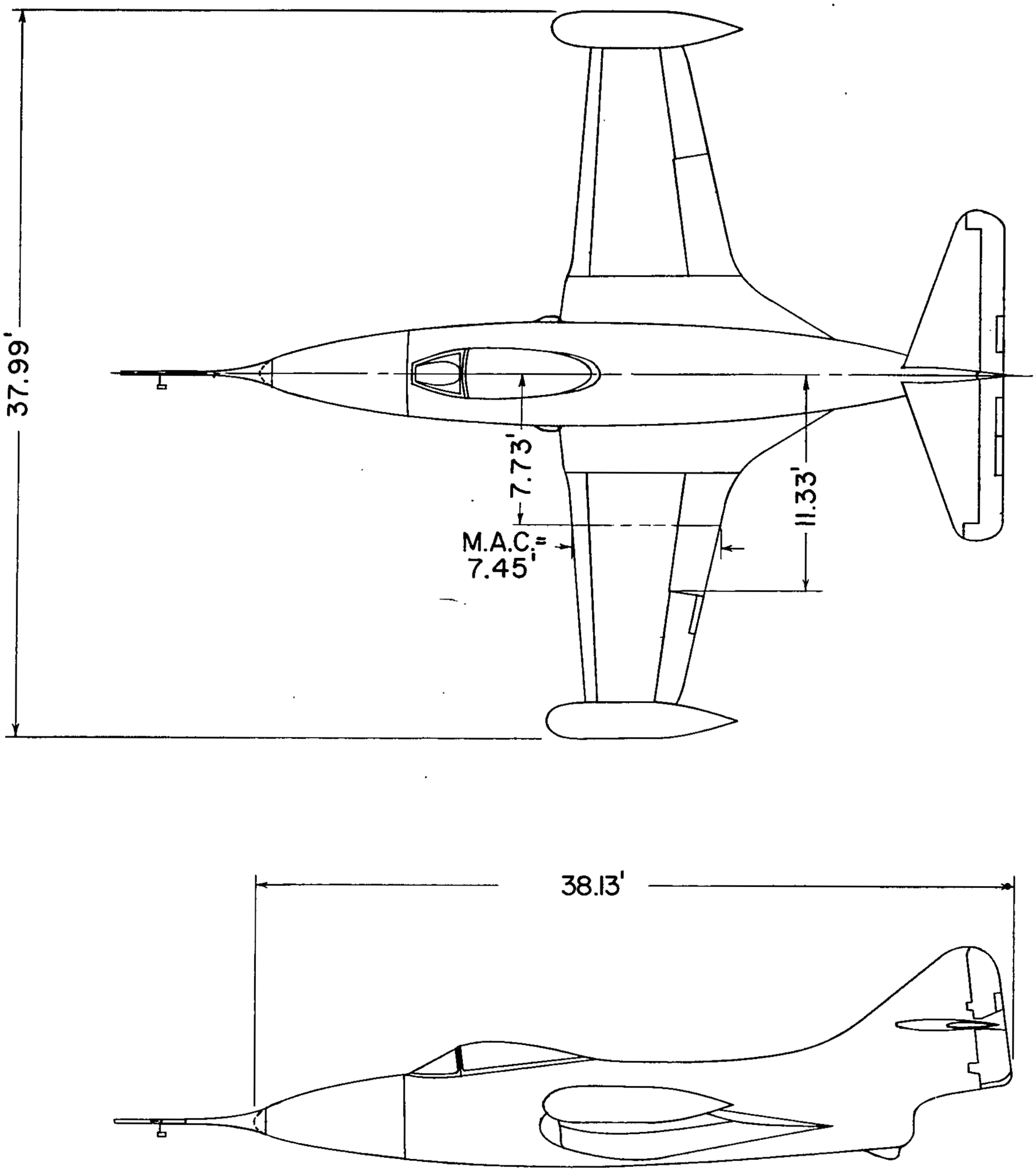
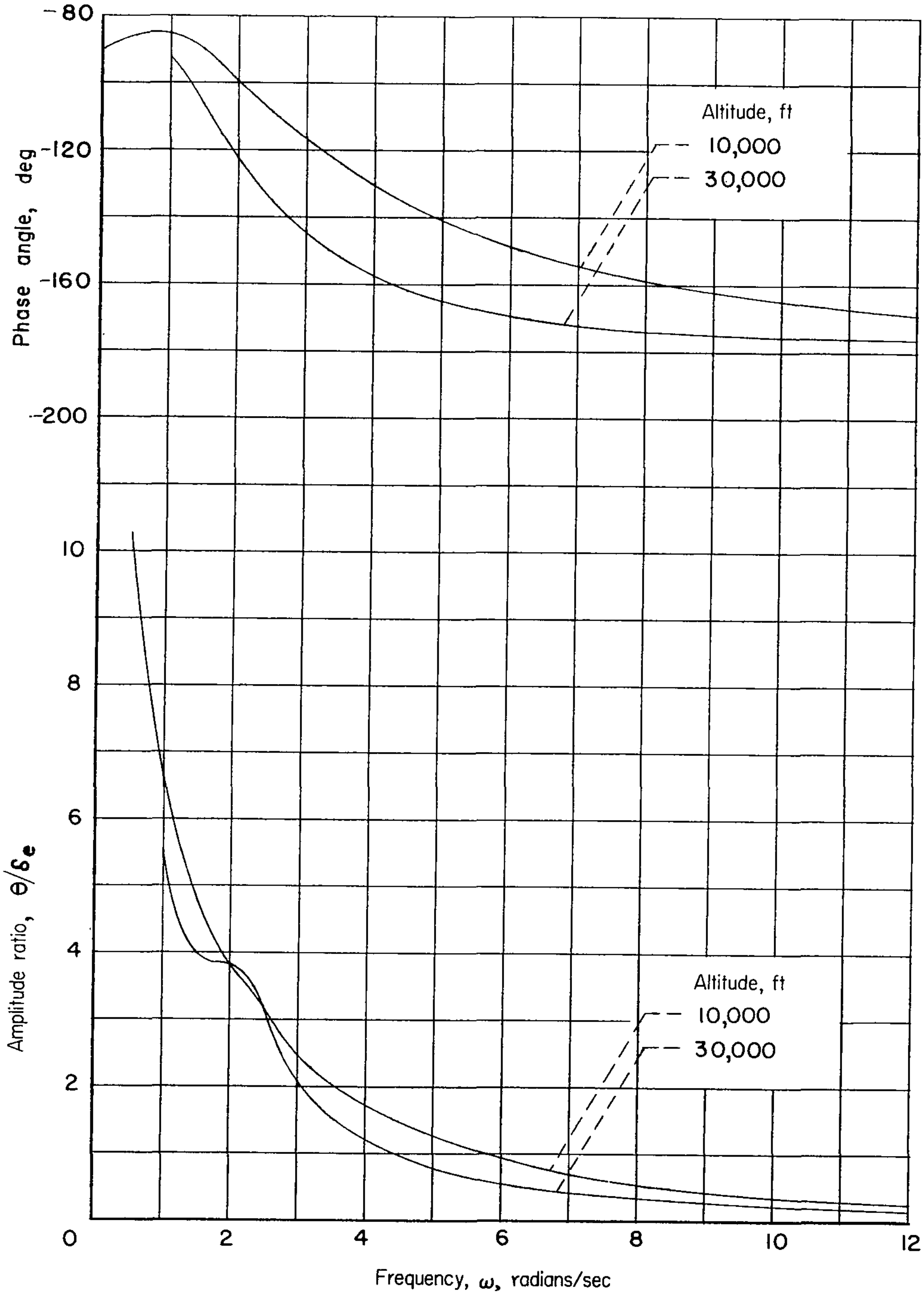
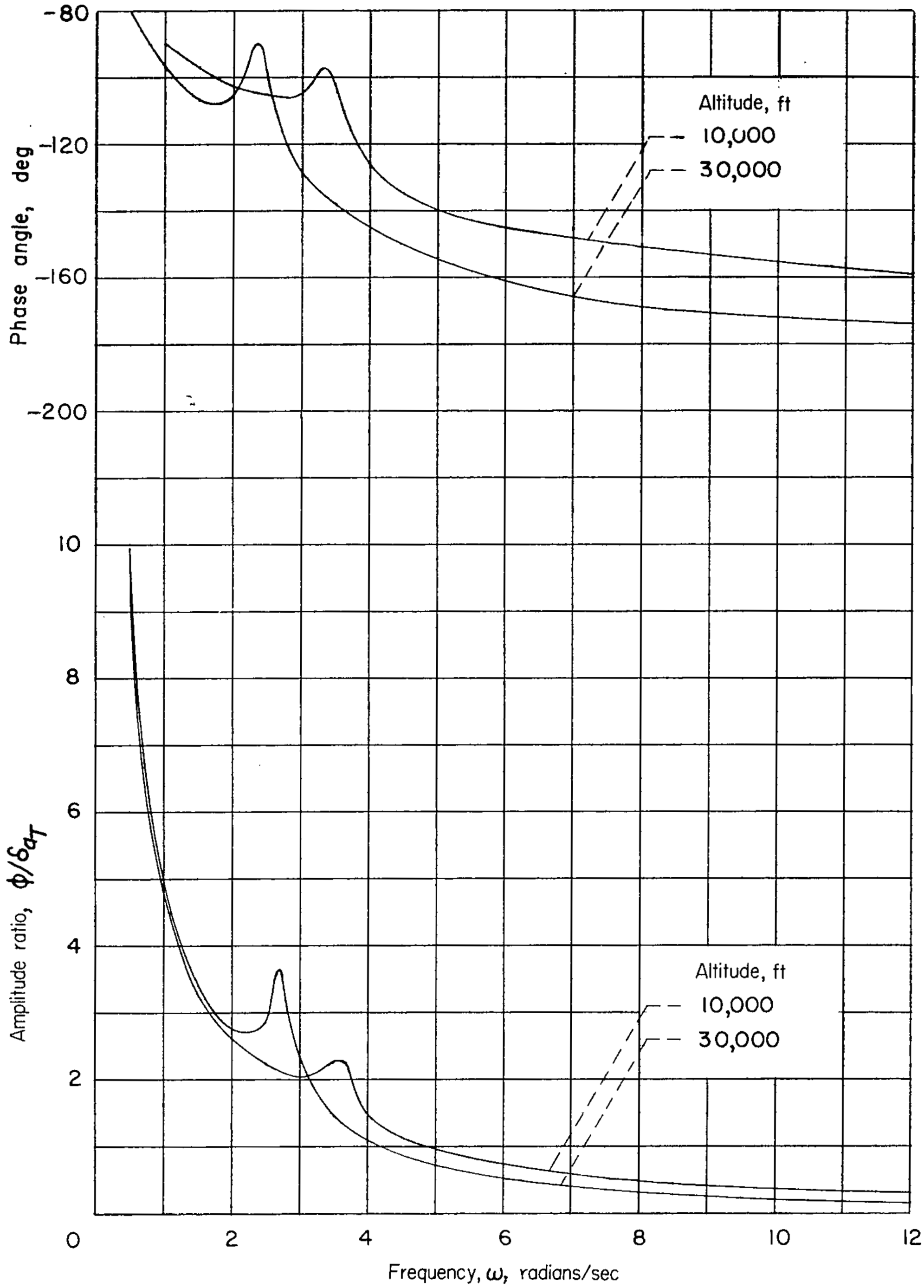


Figure 2.- Two-view drawing of Grumman F9F-2 airplane.



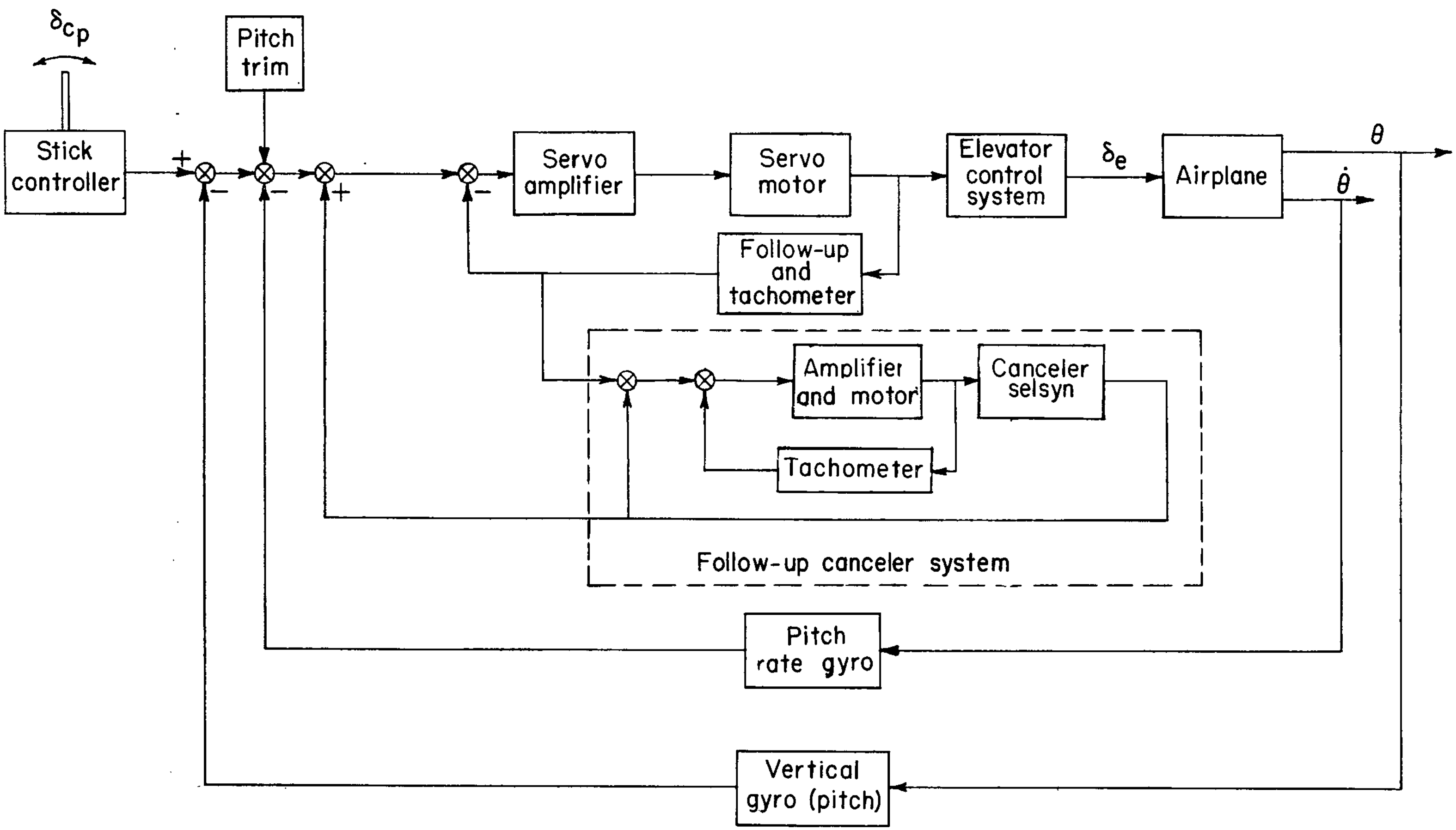
(a) Pitch response, center of gravity at 28.5 percent mean aerodynamic chord.

Figure 3.- Frequency-response characteristics in pitch and roll for the airplane alone. $M = 0.6$, $h_p = 10{,}000$ and $30{,}000$ feet.



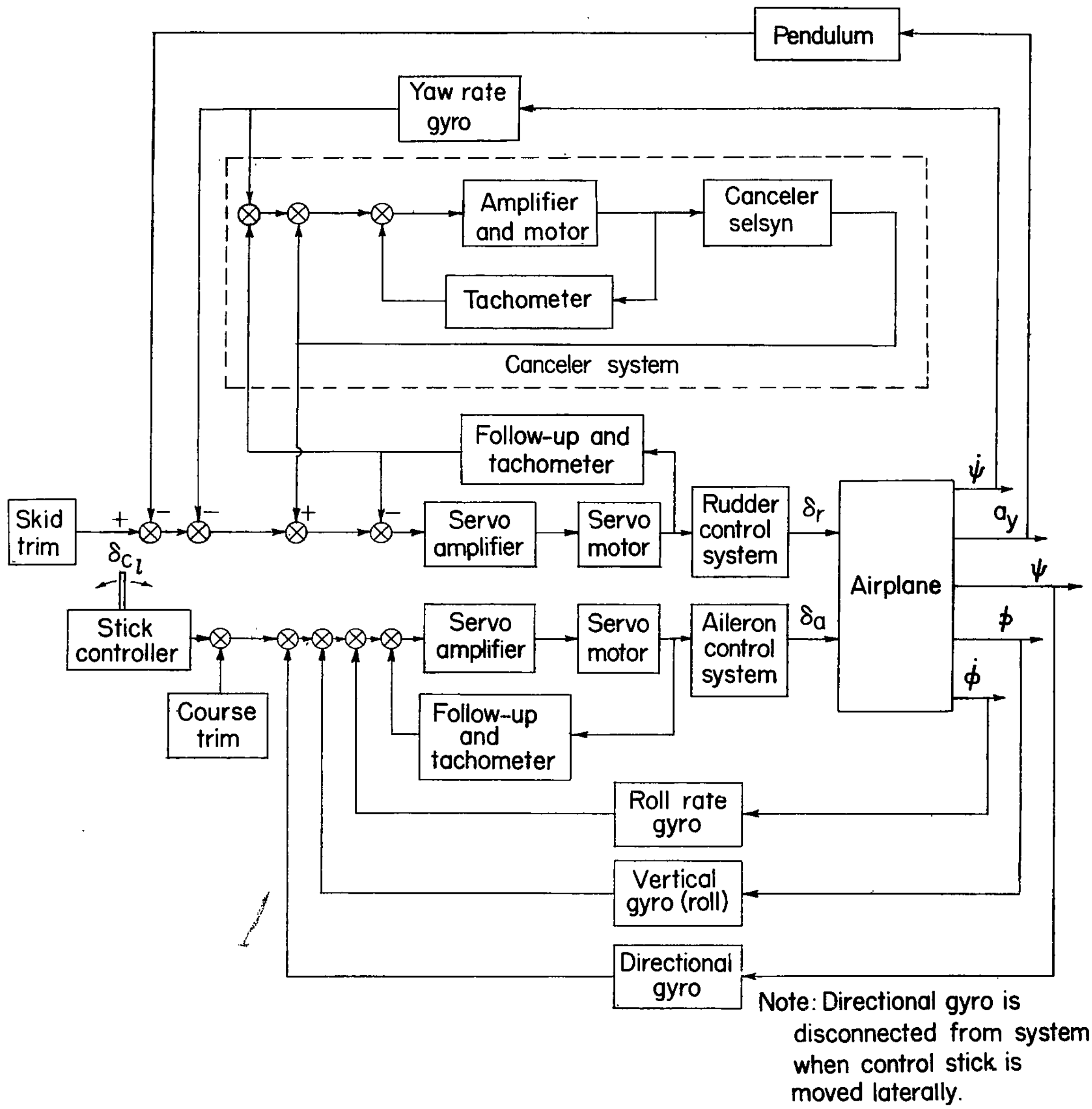
(b) Roll response.

Figure 3.- Concluded.



(a) Pitch channel.

Figure 4.- Block diagrams of airplane—automatic-pilot combination.



(b) Roll and yaw channels.

Figure 4.- Concluded.

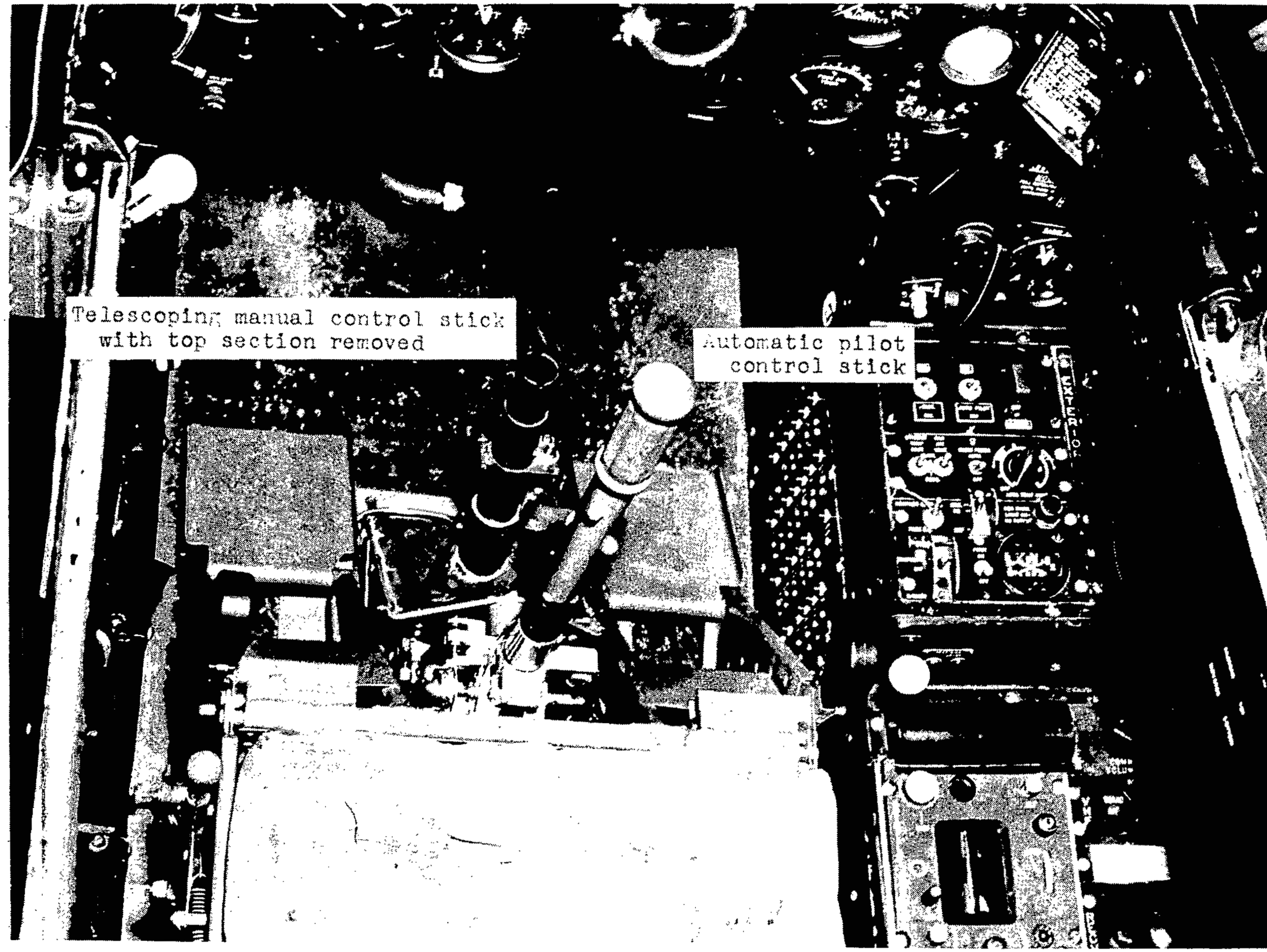


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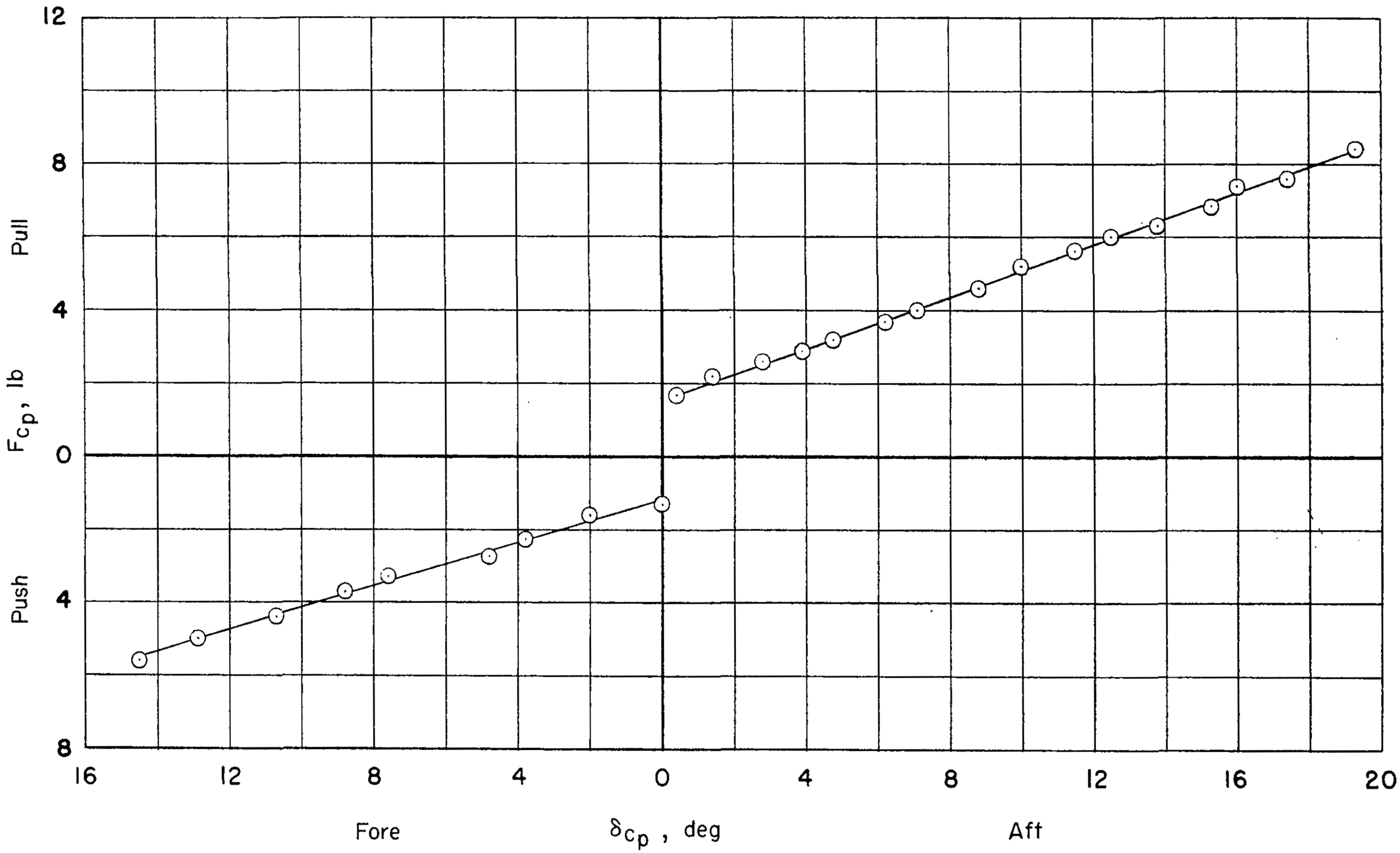
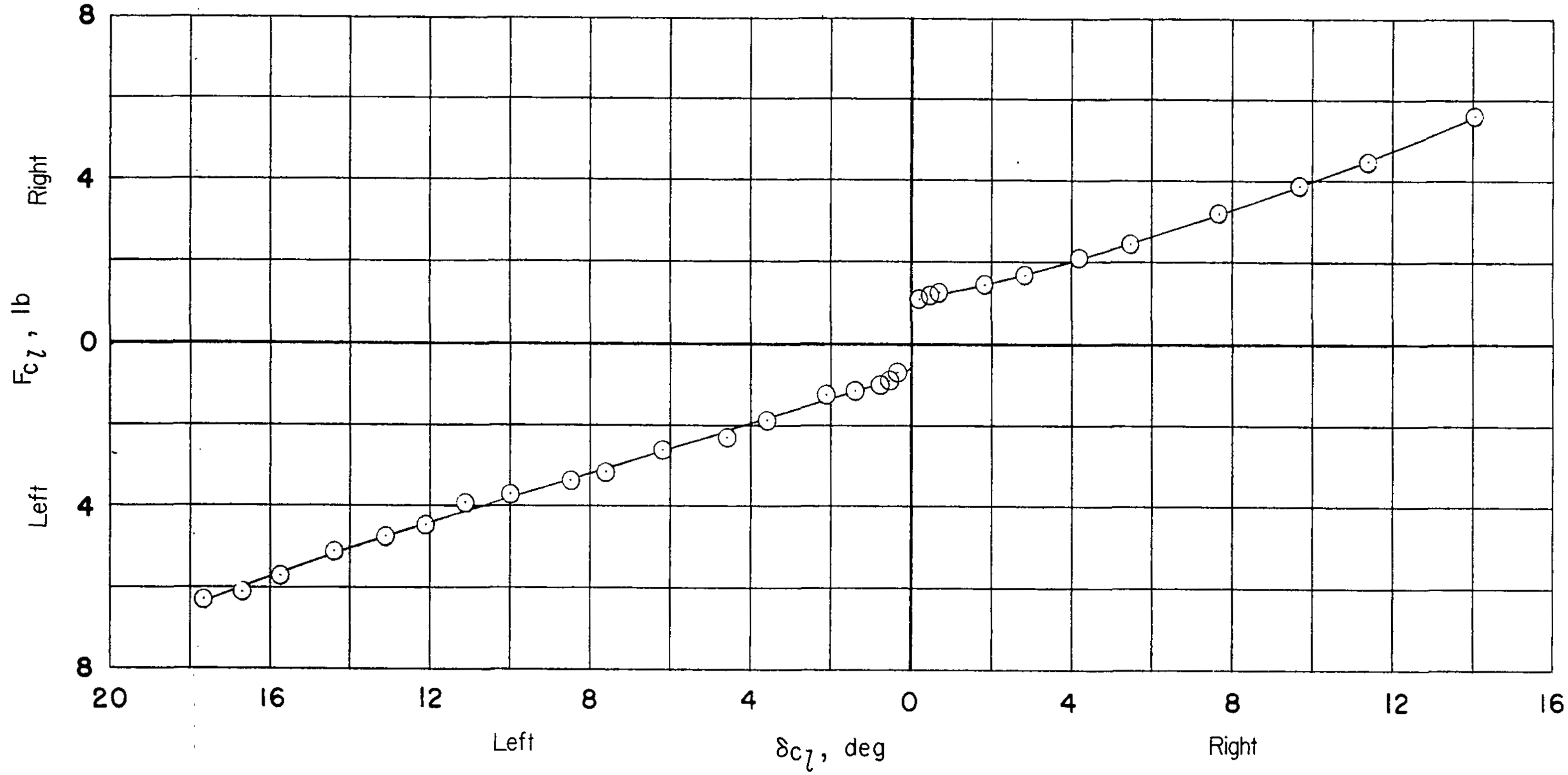
Figure 5.- Top view of airplane cockpit showing automatic-pilot control-
stick installation.



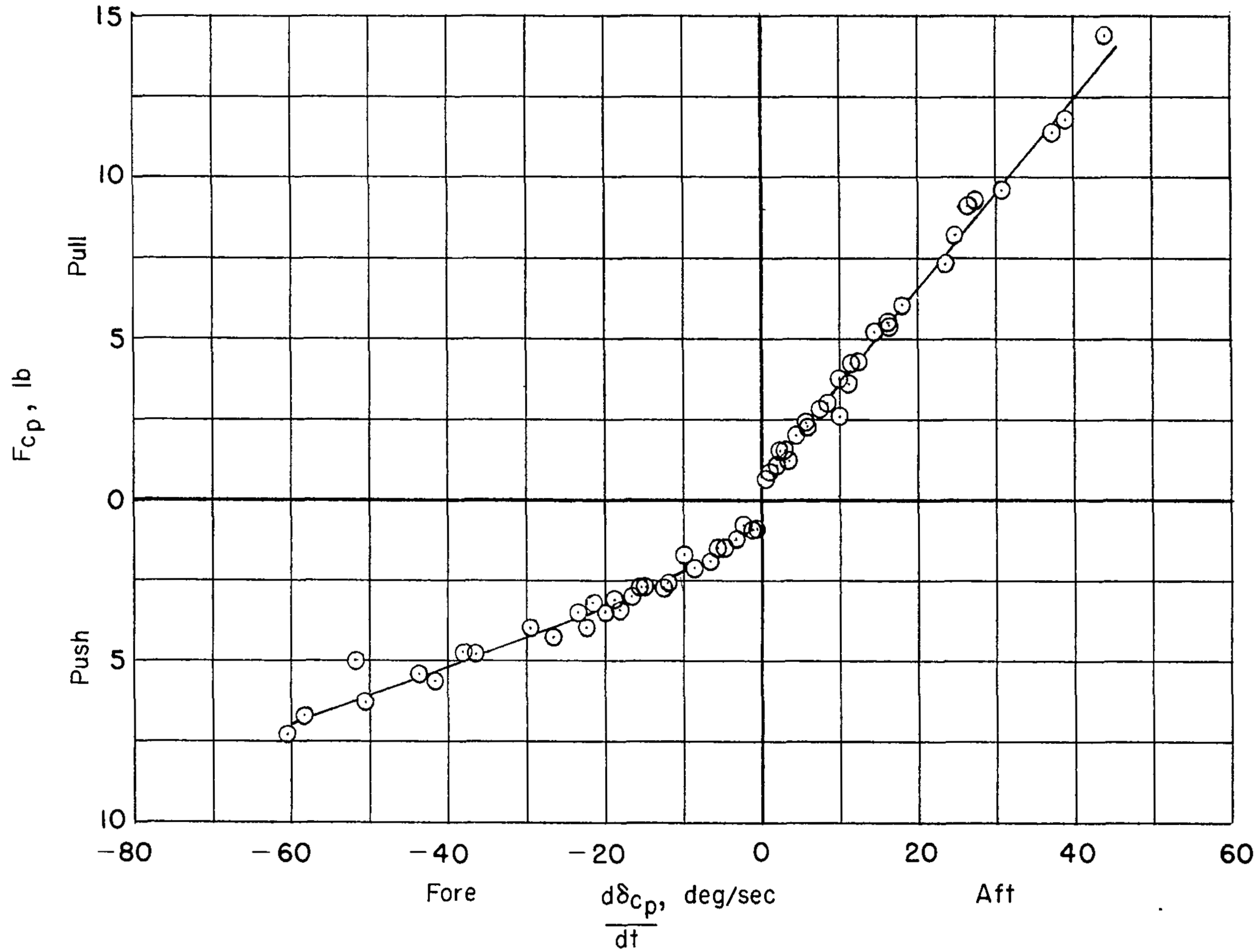
(a) Longitudinal.

Figure 6.- Variation of automatic-pilot control stick force with stick position for spring feel system.



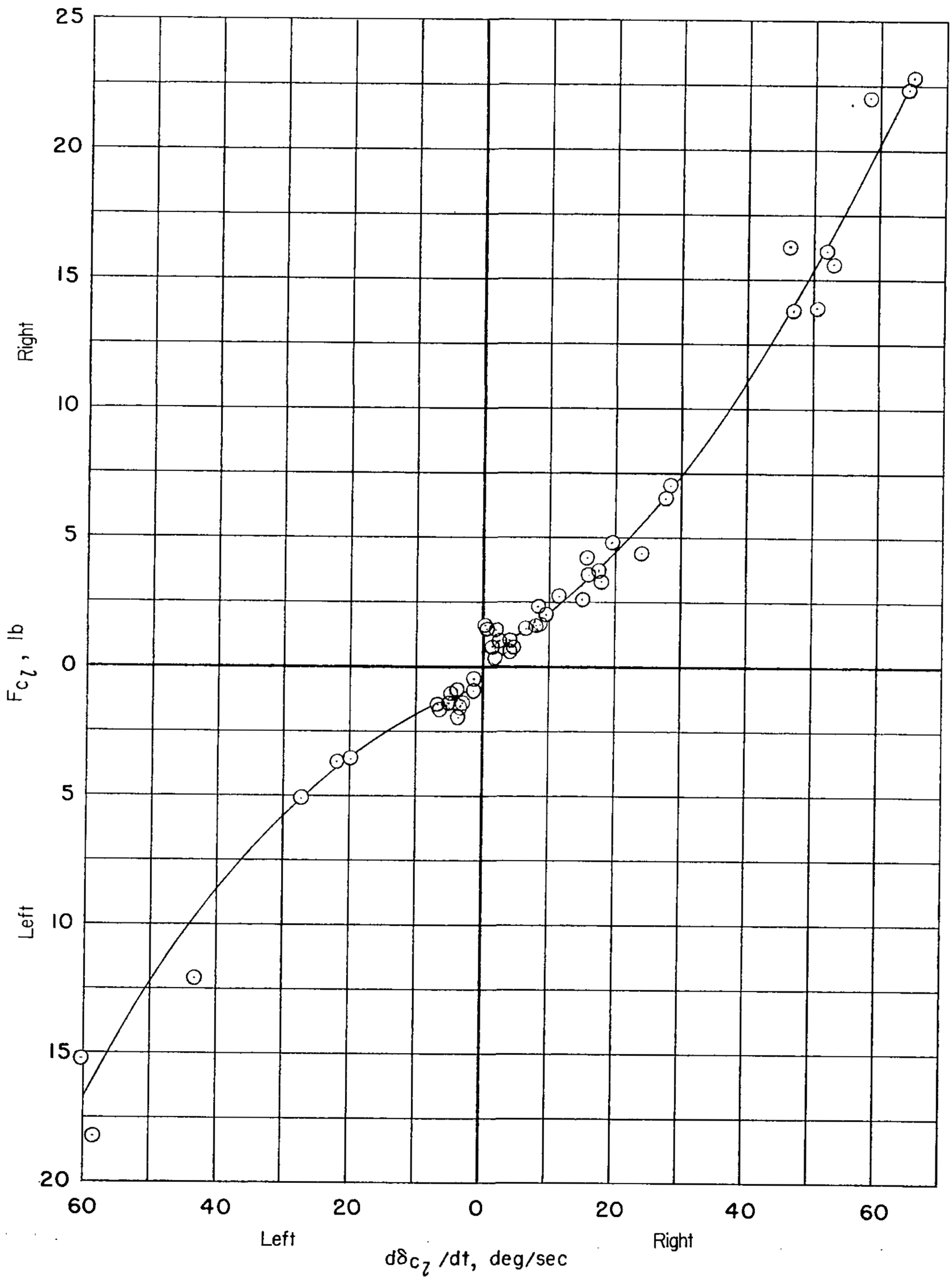
(b) Lateral.

Figure 6.- Concluded.



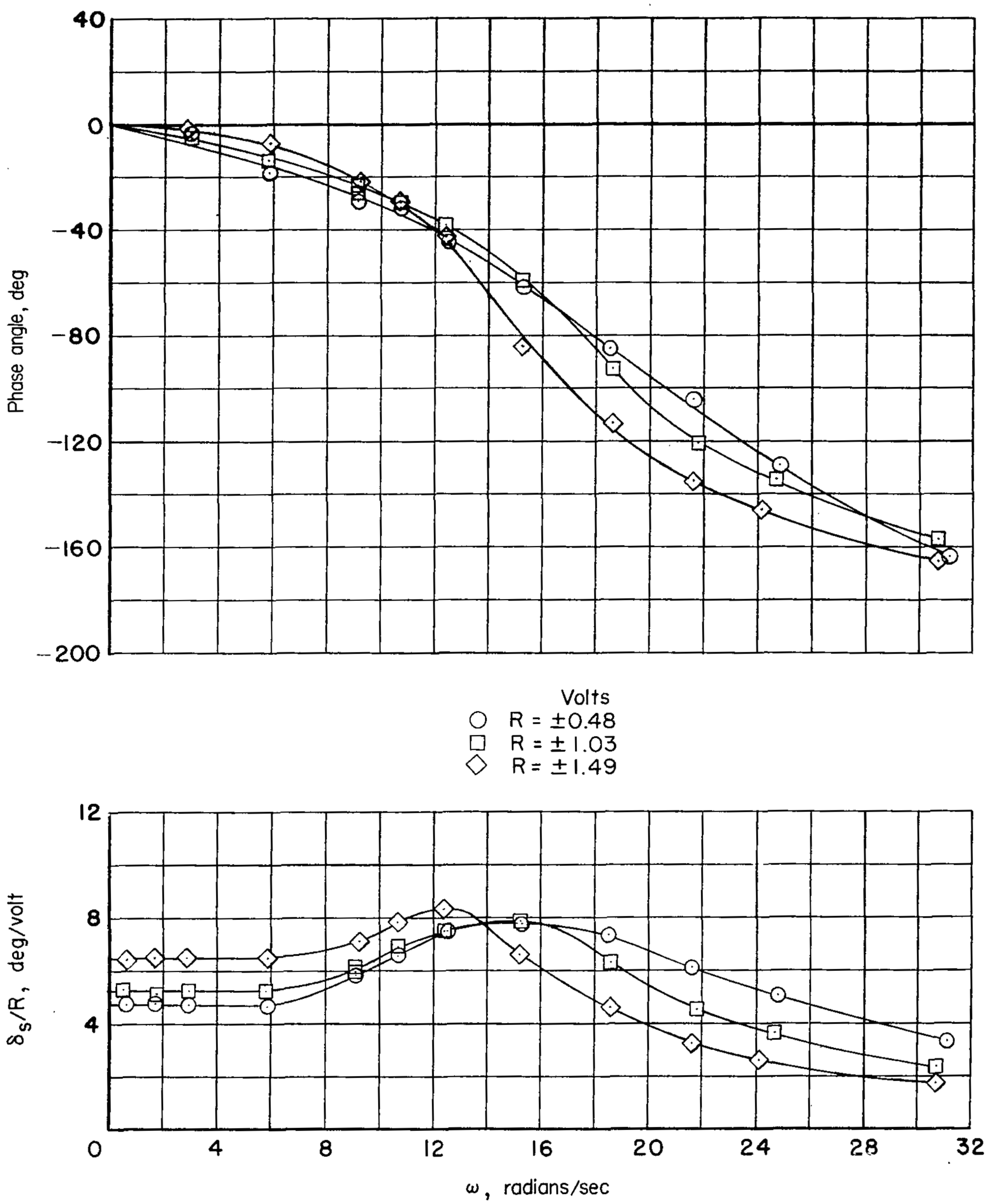
(a) Longitudinal.

Figure 7.- Variation of automatic-pilot control stick force with rate of stick deflection for damper feel system.



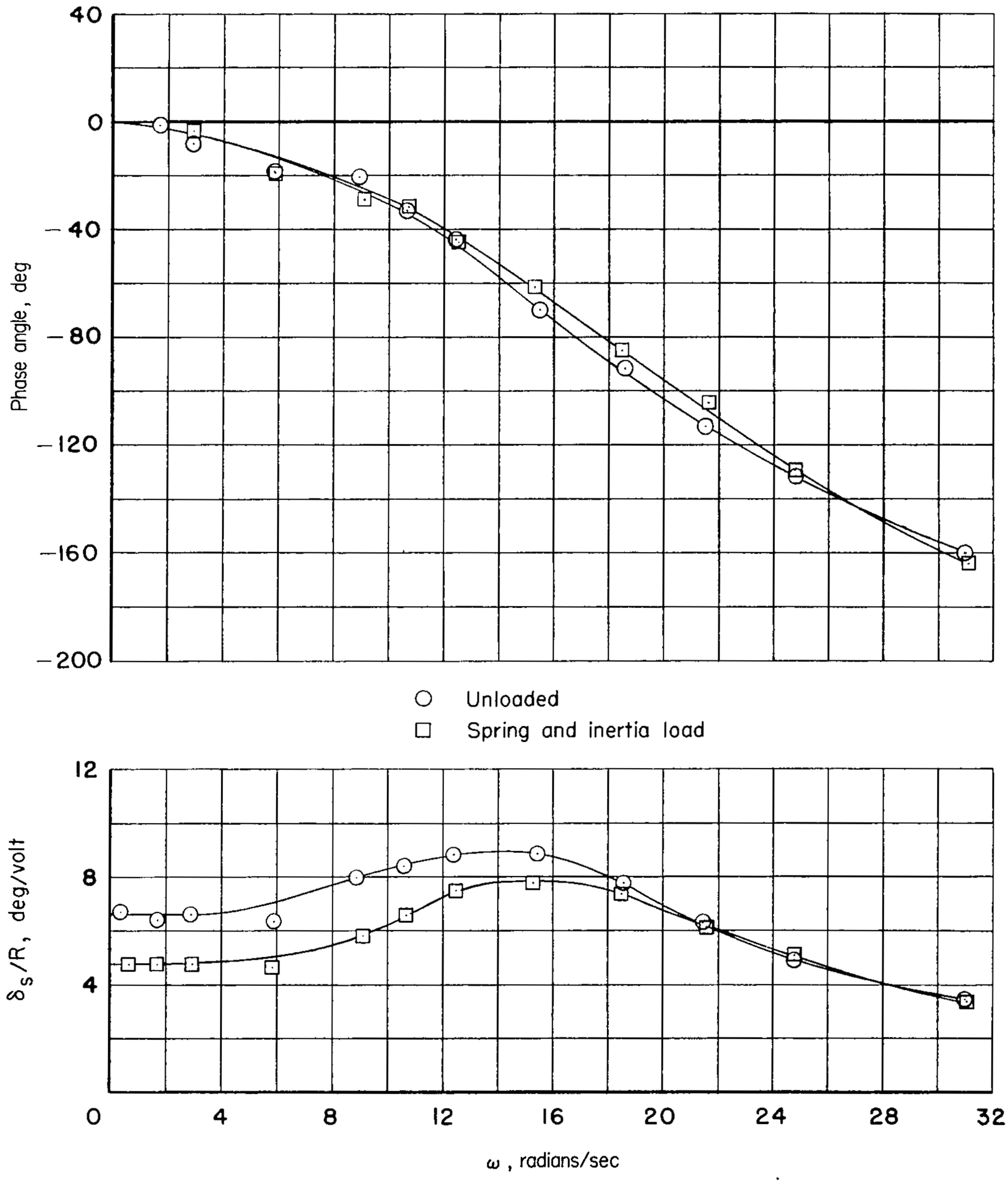
(b) Lateral.

Figure 7.- Concluded.



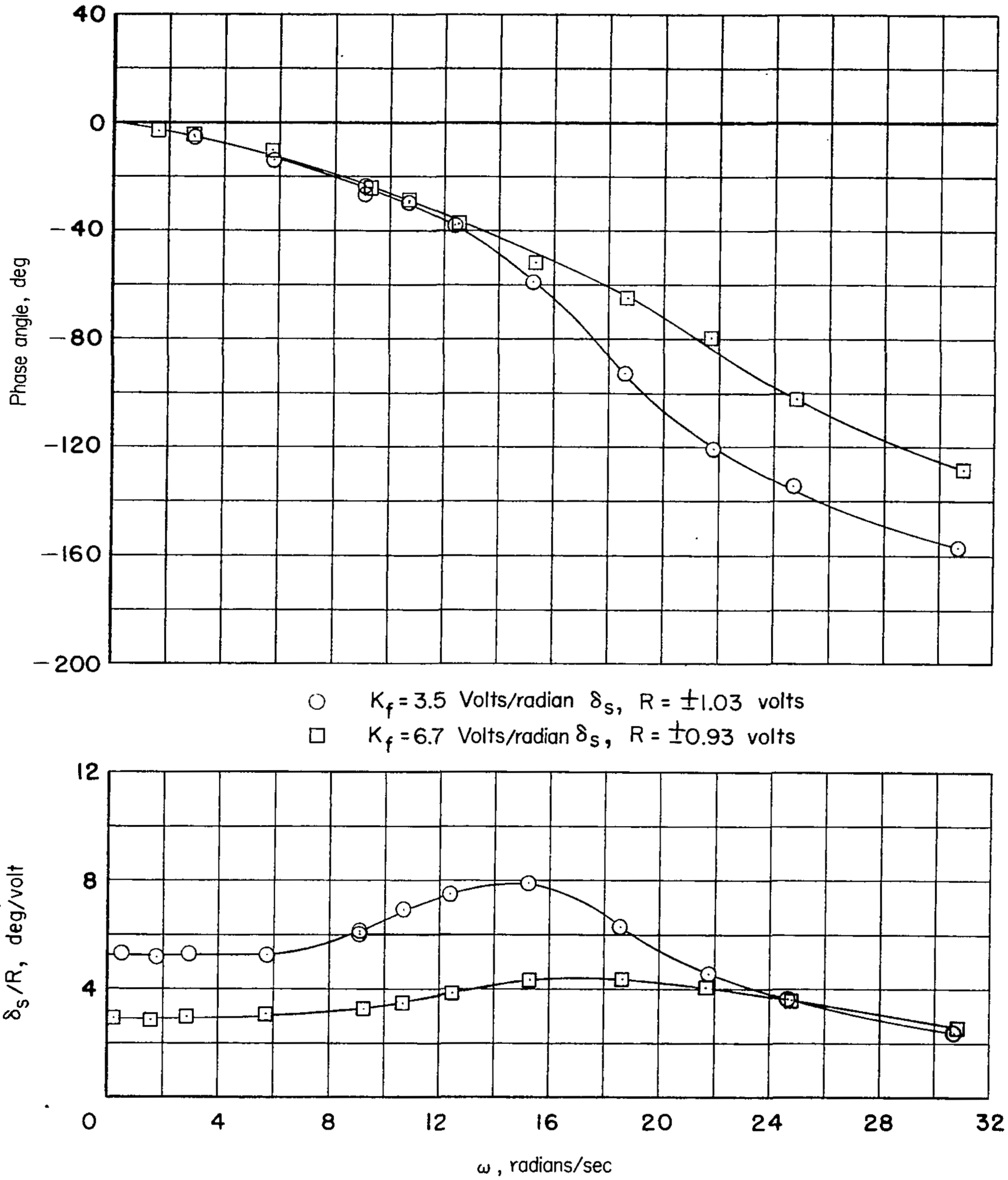
(a) Effect of amplitude of input, servo loaded, moment of inertia about servo output shaft = 0.02 slug-ft^2 (0.5 slug-ft^2 about elevator hinge line), spring rate = 12 in-lb servo torque per degree δ_s (60 in-lb servo torque per degree δ_e), and feedback gain (K_f) = 3.5 volts per radian δ_s.

Figure 8.- Frequency-response characteristics of automatic-pilot servo system.



(b) Effect of inertia and spring load with feedback gain = 3.5 volts per radian δ_s, input signal = ±0.48 volt, moment of inertia about servo motor output shaft = 0.02 slug-ft^2 (0.5 slug-ft^2 about elevator hinge line), spring rate = 12 in-lb servo torque per degree δ_s (60 in-lb servo torque per degree δ_e).

Figure 8.- Continued.



(c) Effect of feedback gain, servo loaded, moment of inertia = 0.02 slug-ft^2, spring rate = 12 in-lb per degree δ_s.

Figure 8.- Concluded.

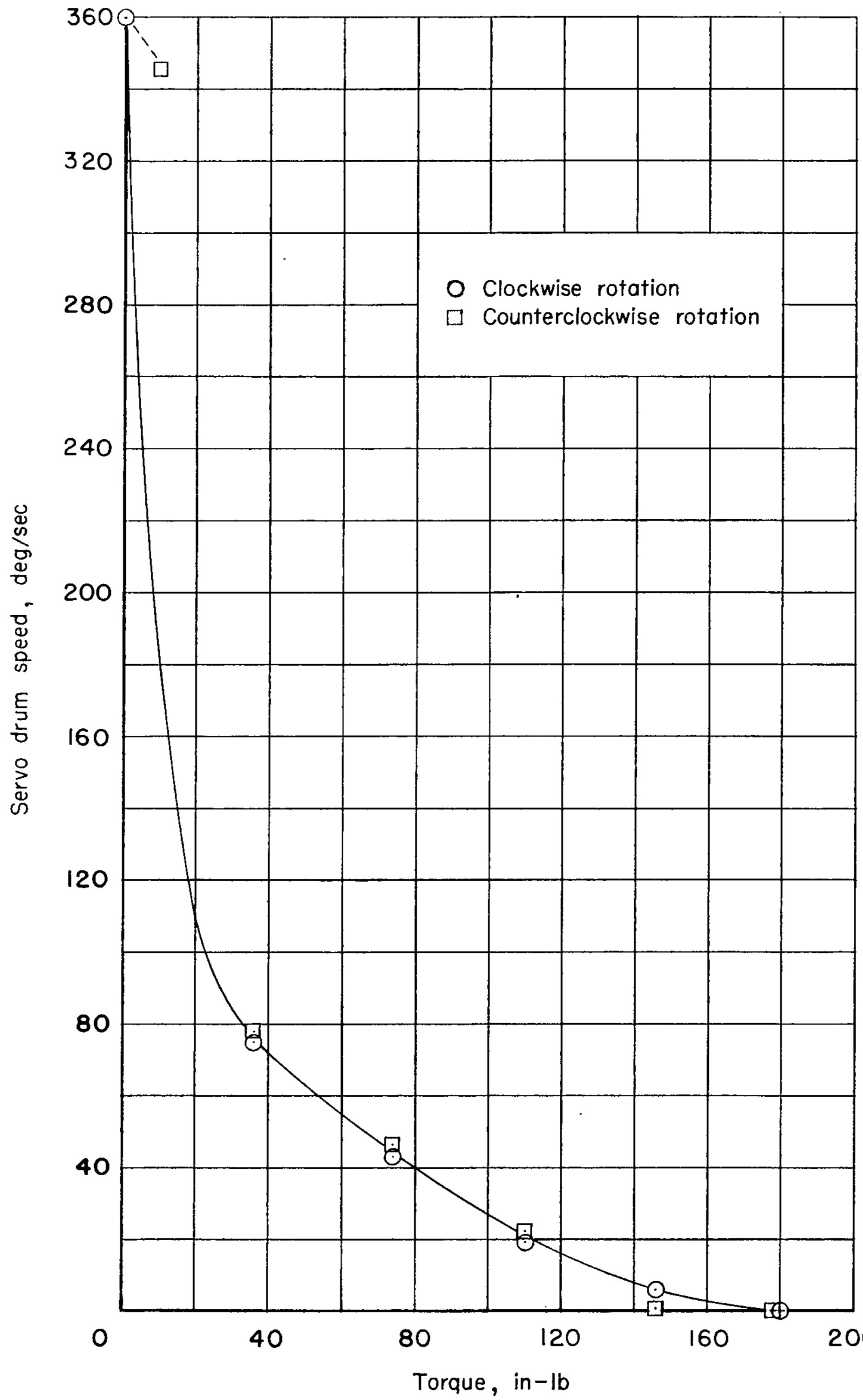


Figure 9.- Speed-torque characteristics of automatic-pilot servo.

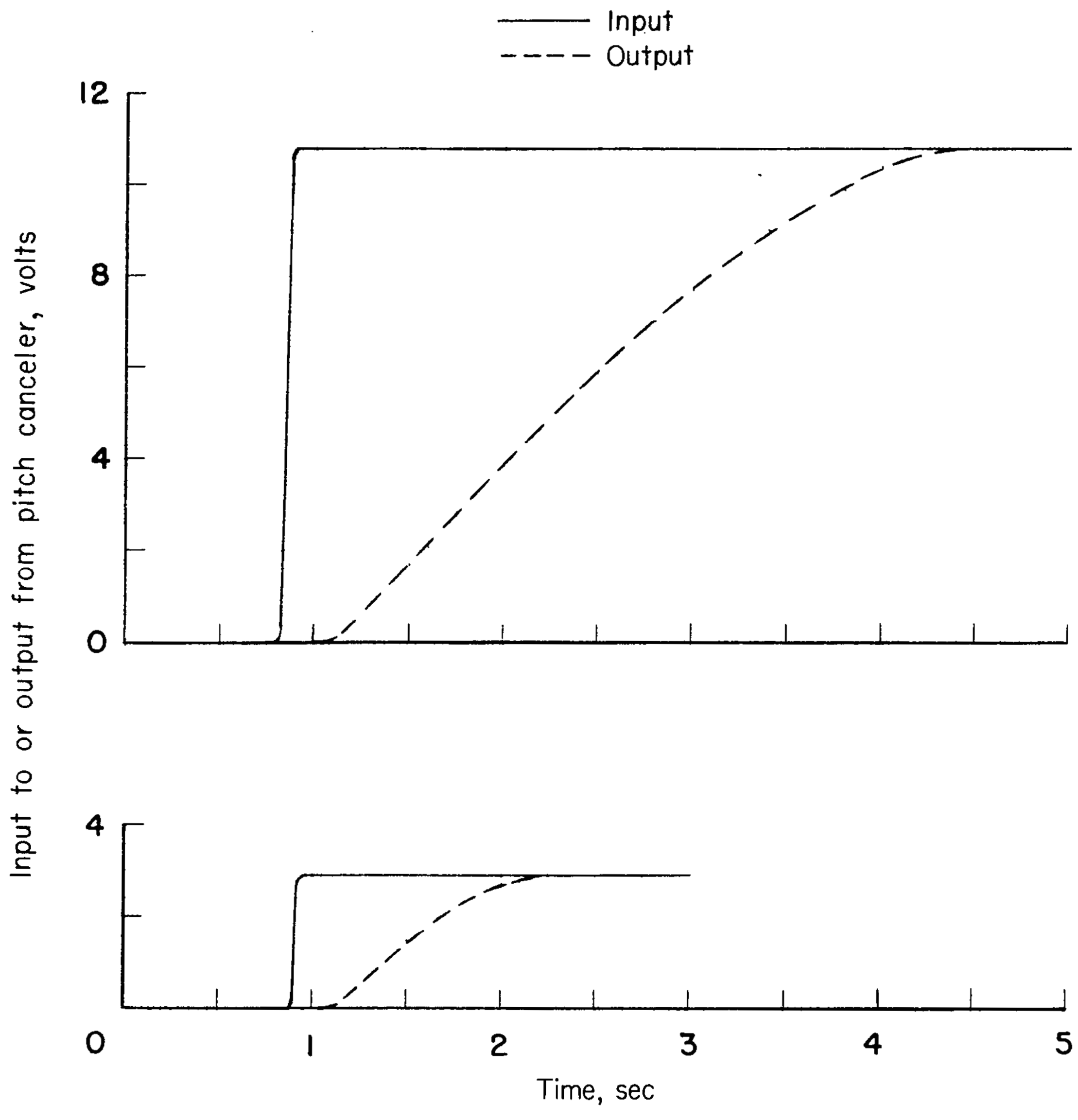
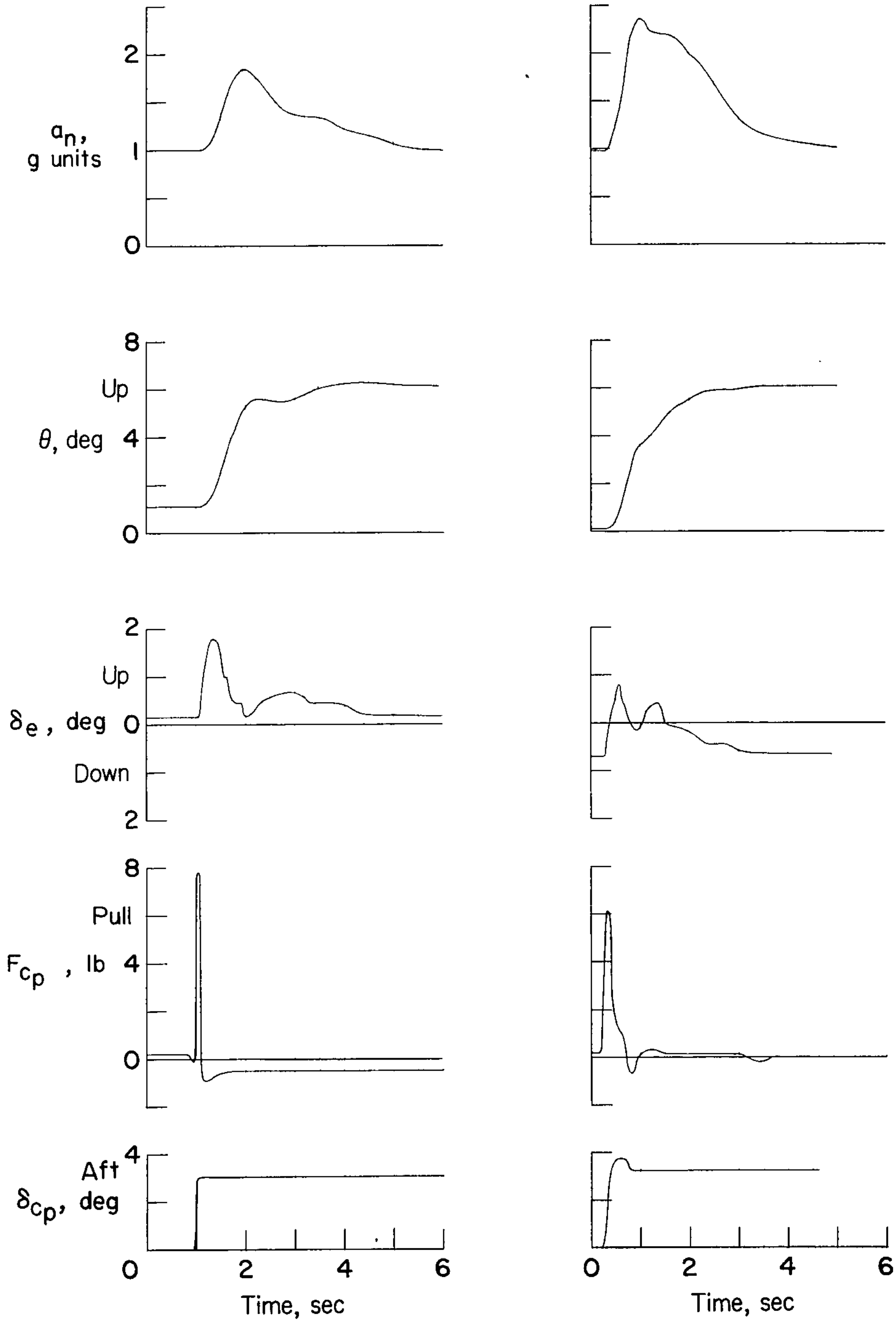


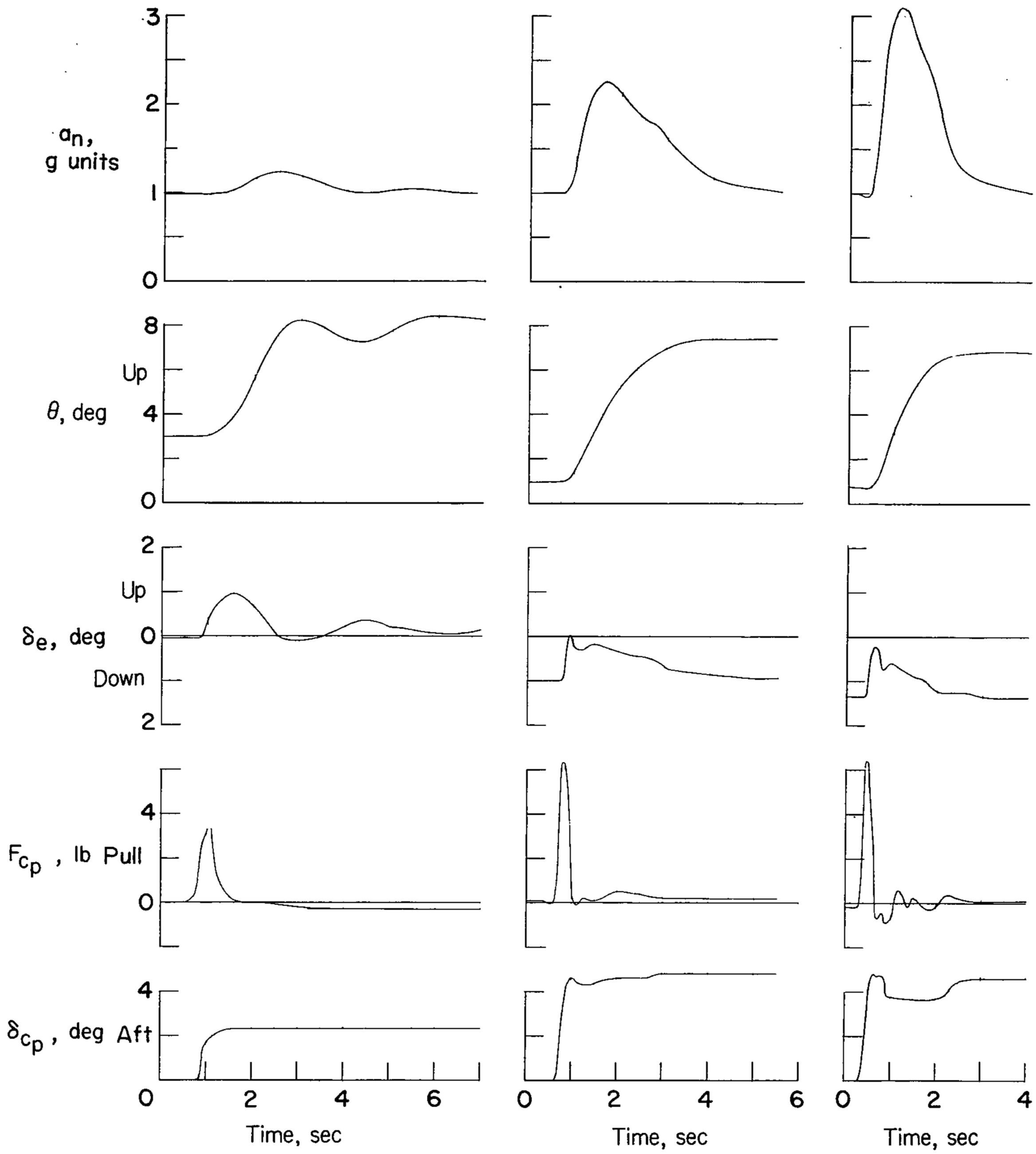
Figure 10.- Transient-response characteristics of pitch canceler system for two amplitudes of input signal.



(a) Clean condition, power for level flight, M = 0.6, h_p = 30,000 feet, K_θ = 15.5 volts/radian, $K_{\dot\theta}$ = 9.2 volts/radian/sec, K_{f_e} = 3.5 volts/radian.

(b) Clean condition, power for level flight, M = 0.76, h_p = 30,000 feet, K_θ = 15.5 volts/radian, $K_{\dot\theta}$ = 9.2 volts/radian/sec, K_{f_e} = 3.5 volts/radian.

Figure 11.- Transient-response characteristics in pitch of airplane—automatic-pilot combination with damper feel system.



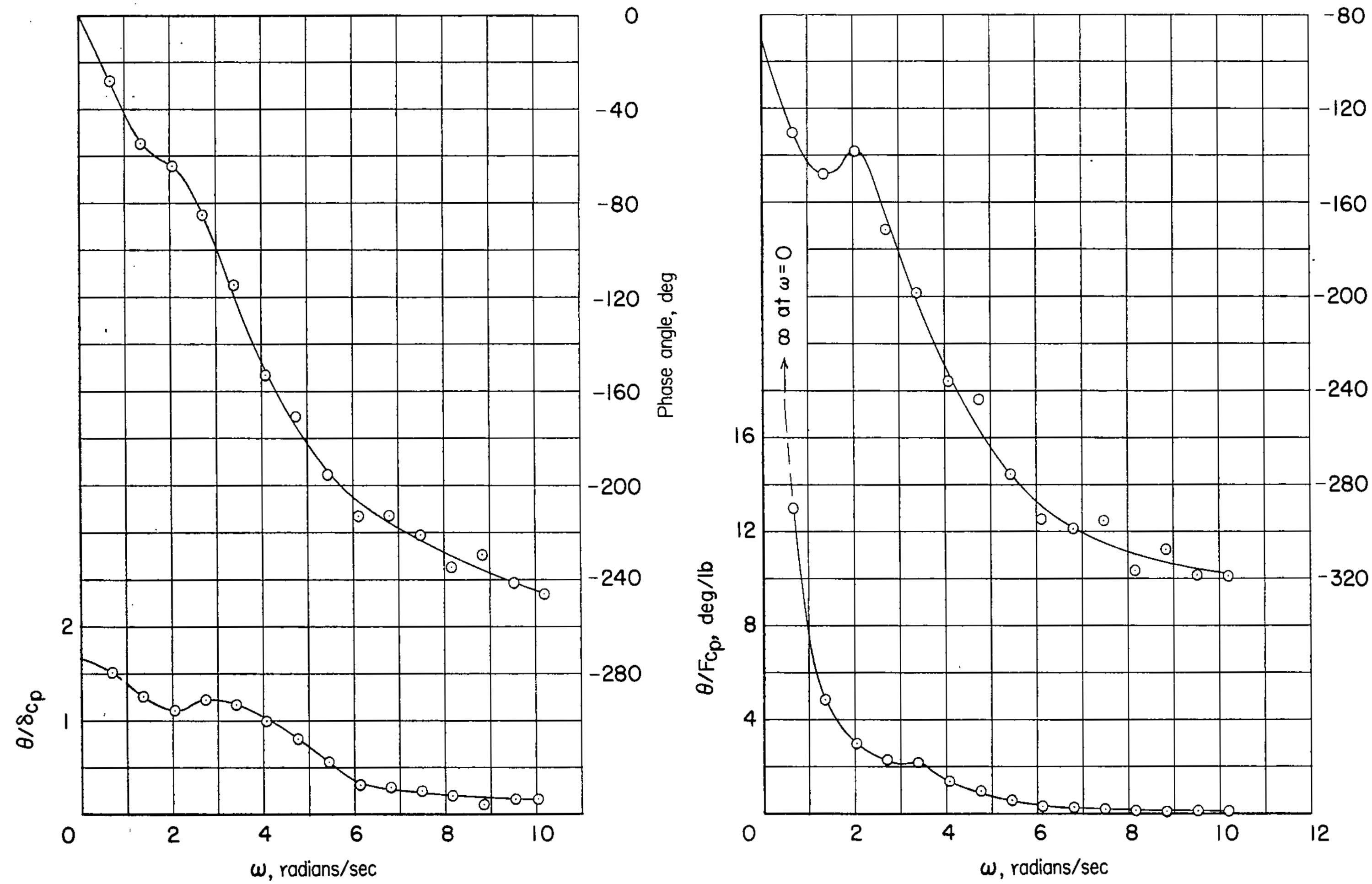
(c) Power approach condition, power for level flight, V_i = 125 knots, h_p = 5,000 feet, K_θ = 15.5 volts/radian, $K_{\dot{\theta}}$ = 11.6 volts/radian/sec, K_{f_e} = 7.0 volts/radian.

(d) Clean condition, power for level flight, M = 0.6, h_p = 5,000 feet, K_θ = 15.5 volts/radian, $K_{\dot{\theta}}$ = 11.6 volts/radian/sec, K_{f_e} = 7.0 volts/radian.

(e) Clean condition, power for level flight, M = 0.7, h_p = 5,000 feet, K_θ = 15.5 volts/radian, $K_{\dot{\theta}}$ = 6.5 volts/radian/sec, K_{f_e} = 7.0 volts/radian.

Figure 11.- Concluded.

[redacted]

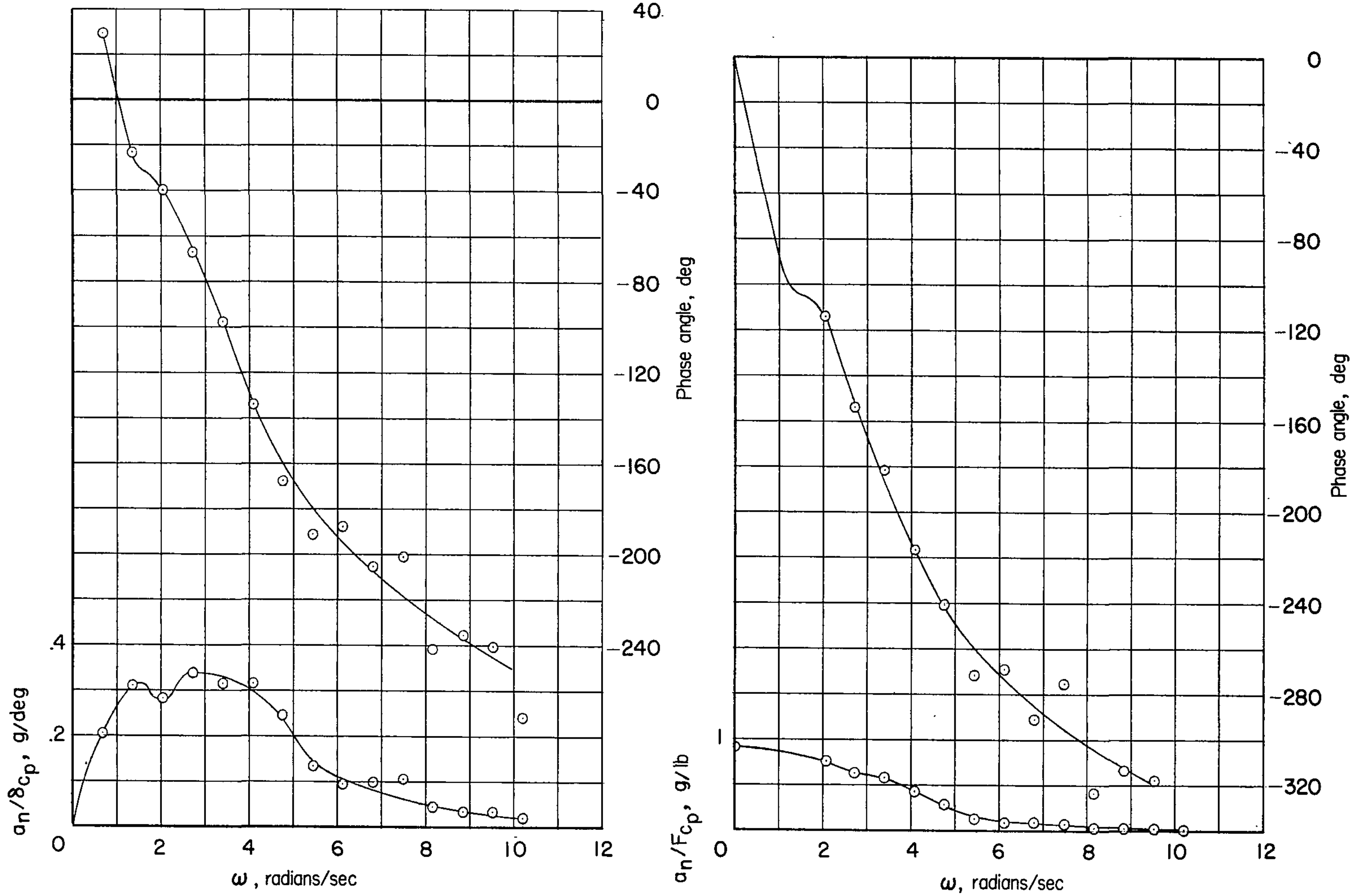


(a) θ/δ_{c_p}.

(b) θ/F_{c_p}.

Figure 12.- Frequency-response characteristics in pitch of airplane—automatic-pilot combination with damper feel system. $M = 0.6$, $h_p = 30,000$ feet, $K_\theta = 15.5$ volts/radian, $K_{\dot{\theta}} = 9.2$ volts/radian/sec, $K_{f_e} = 3.5$ volts/radian.

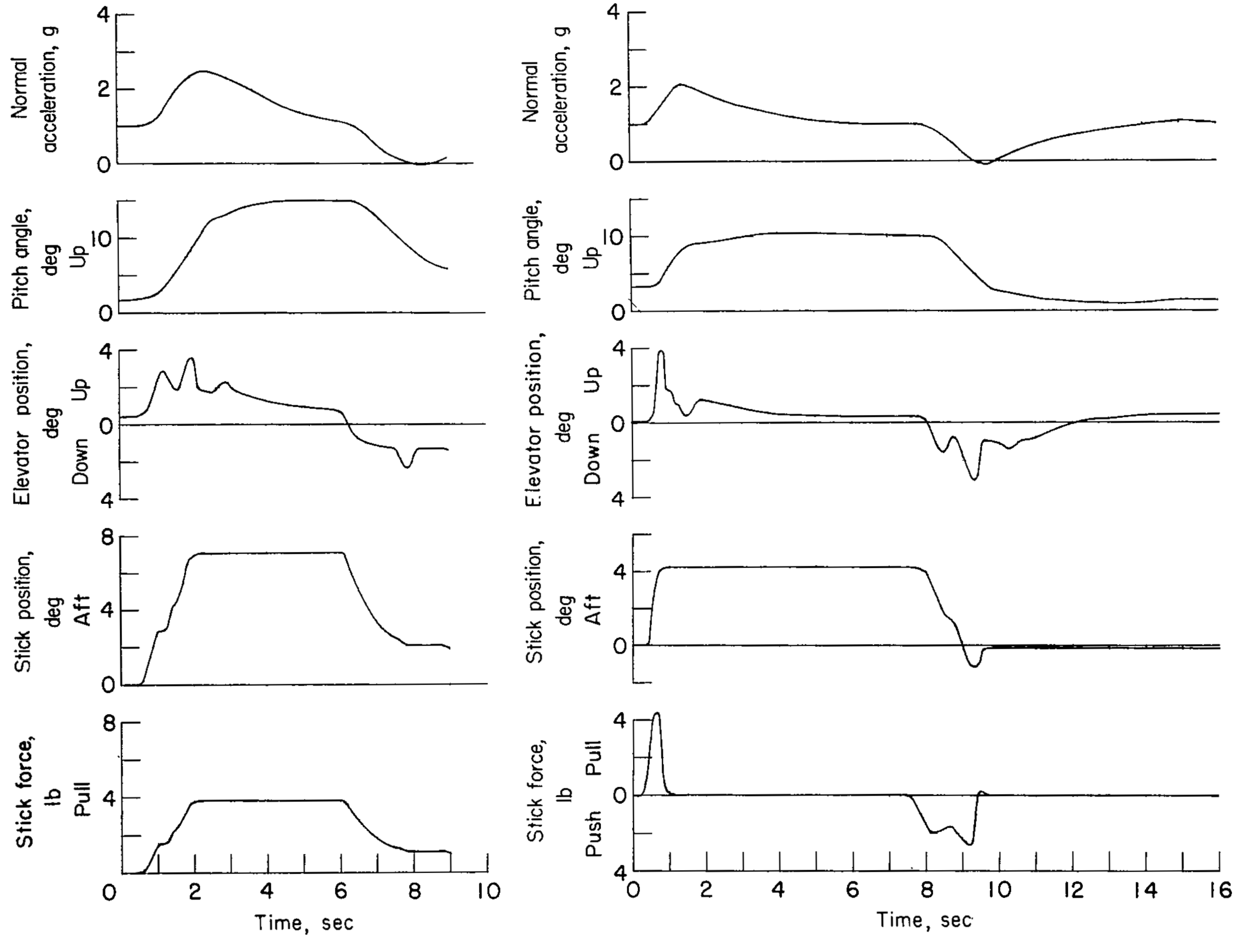
[redacted]



(c) a_n/δ_{c_p}.

(d) a_n/F_{c_p}.

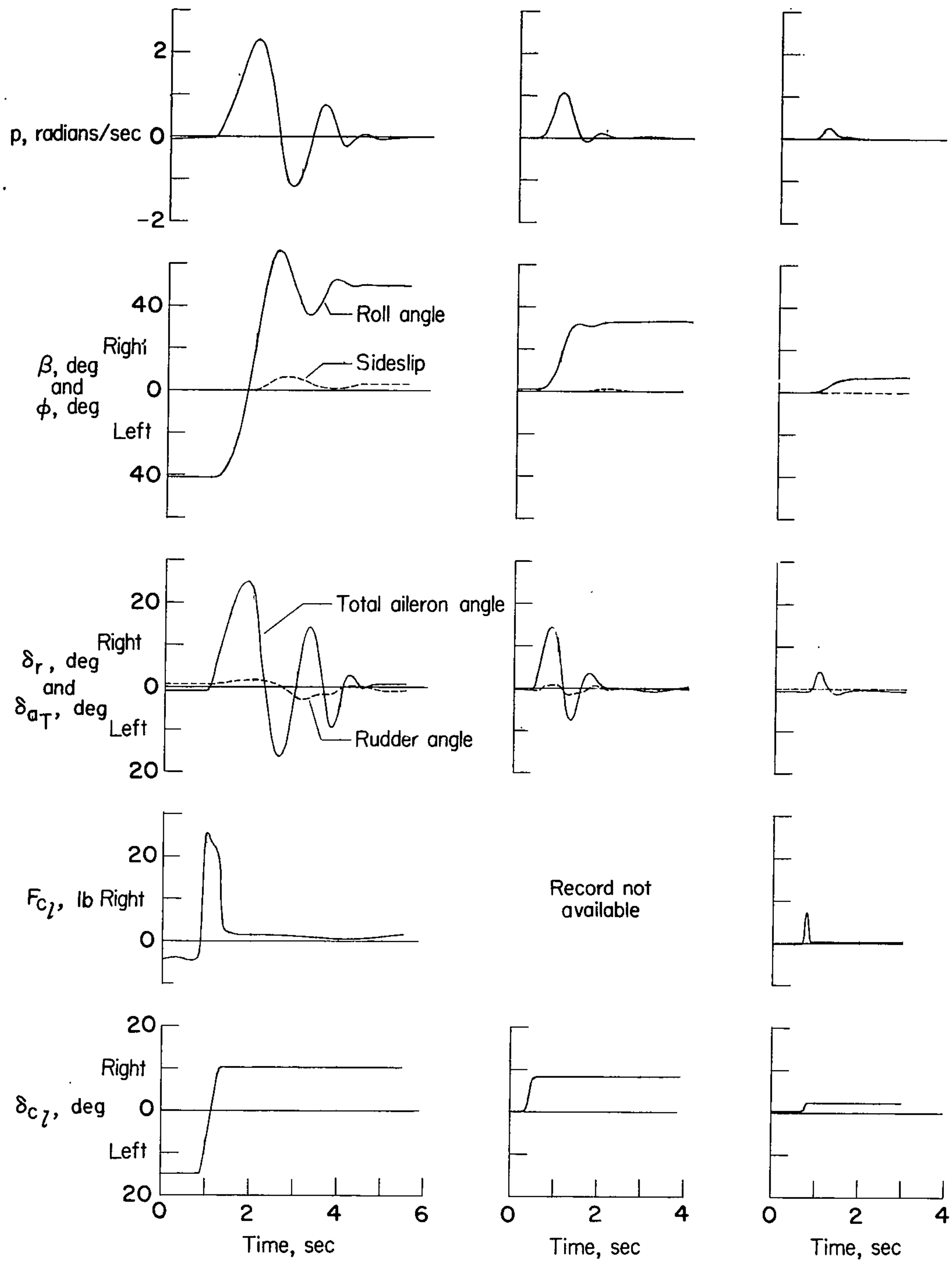
Figure 12.- Concluded.



(a) Spring feel system.

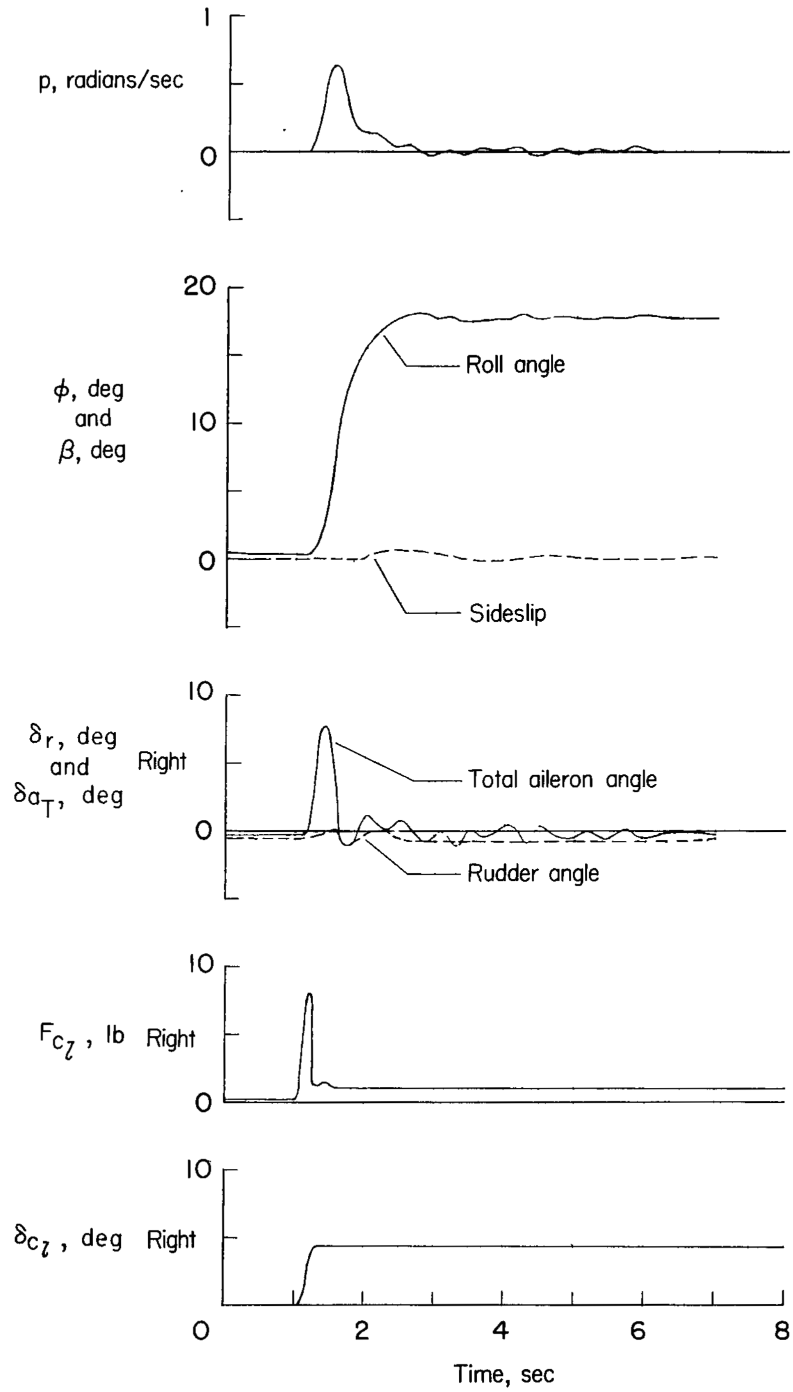
(b) Damper feel system.

Figure 13.- Time histories of pull-up and push-down maneuvers with spring and damper feel systems. $M = 0.6$, $h_p = 30{,}000$ feet, $K_\theta = 15.5$ volts/radian, $K_{\dot{\theta}} = 11.6$ volts/radian/sec, $K_{f_e} = 3.5$ volts/radian.



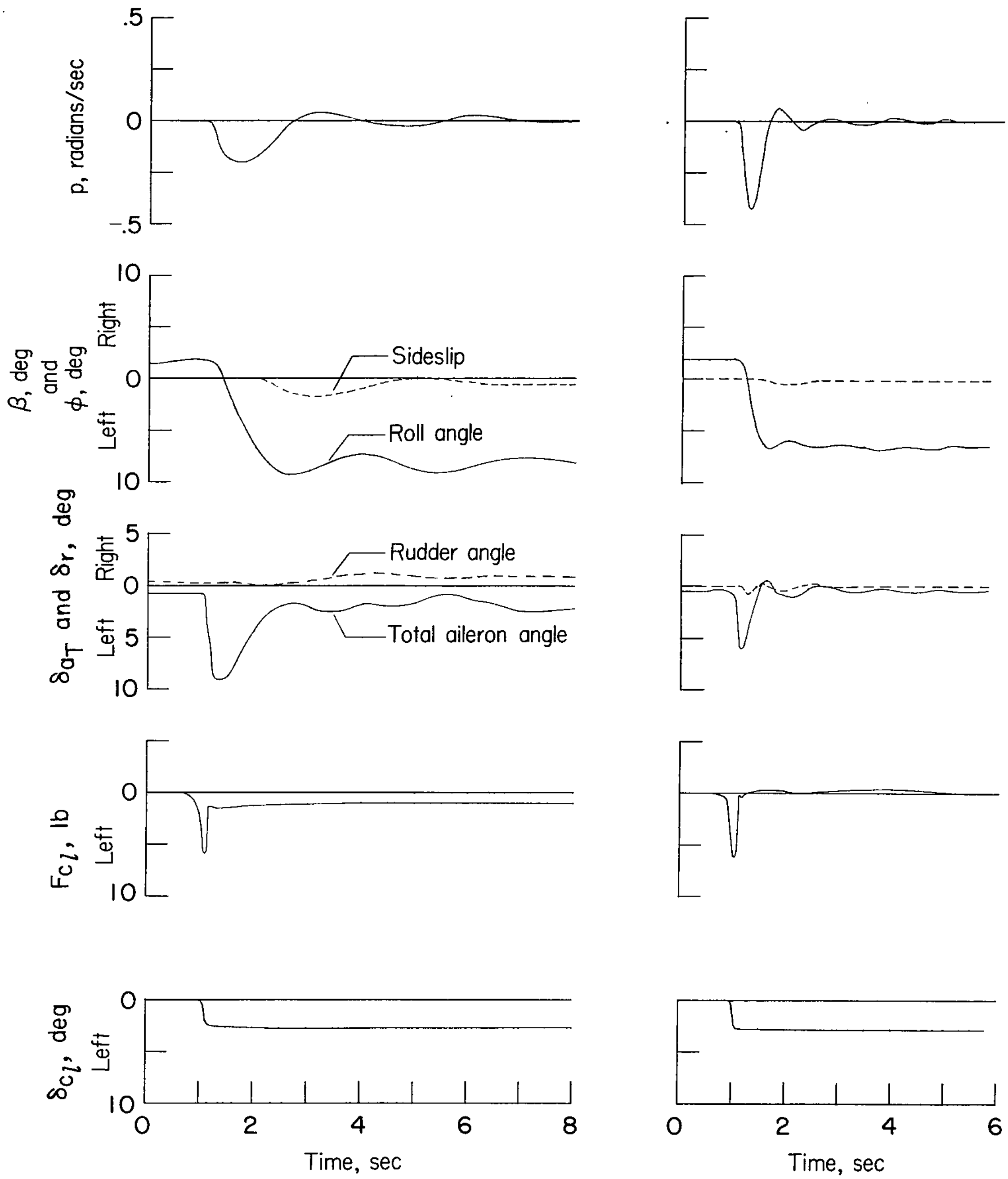
(a) Effect of amplitude, clean condition, power for level flight, $M = 0.6$, $h_p = 30{,}000$ feet, $K_{\phi} = 14.3$ volts/radian, $K_{\dot{\phi}} = 16.7$ volts/radian/sec, $K_{f_a} = 7.0$ volts/radian.

Figure 14.- Transient response characteristics in roll of airplane—automatic-pilot combination with damper-feel system.



(b) Clean condition, power for level flight, $M = 0.7$, $h_p = 30{,}000$ feet, $K_\phi = 14.3$ volts/radian, $K_{\dot{\phi}} = 16.7$ volts/radian/sec, $K_{f_a} = 7.0$ volts/radian.

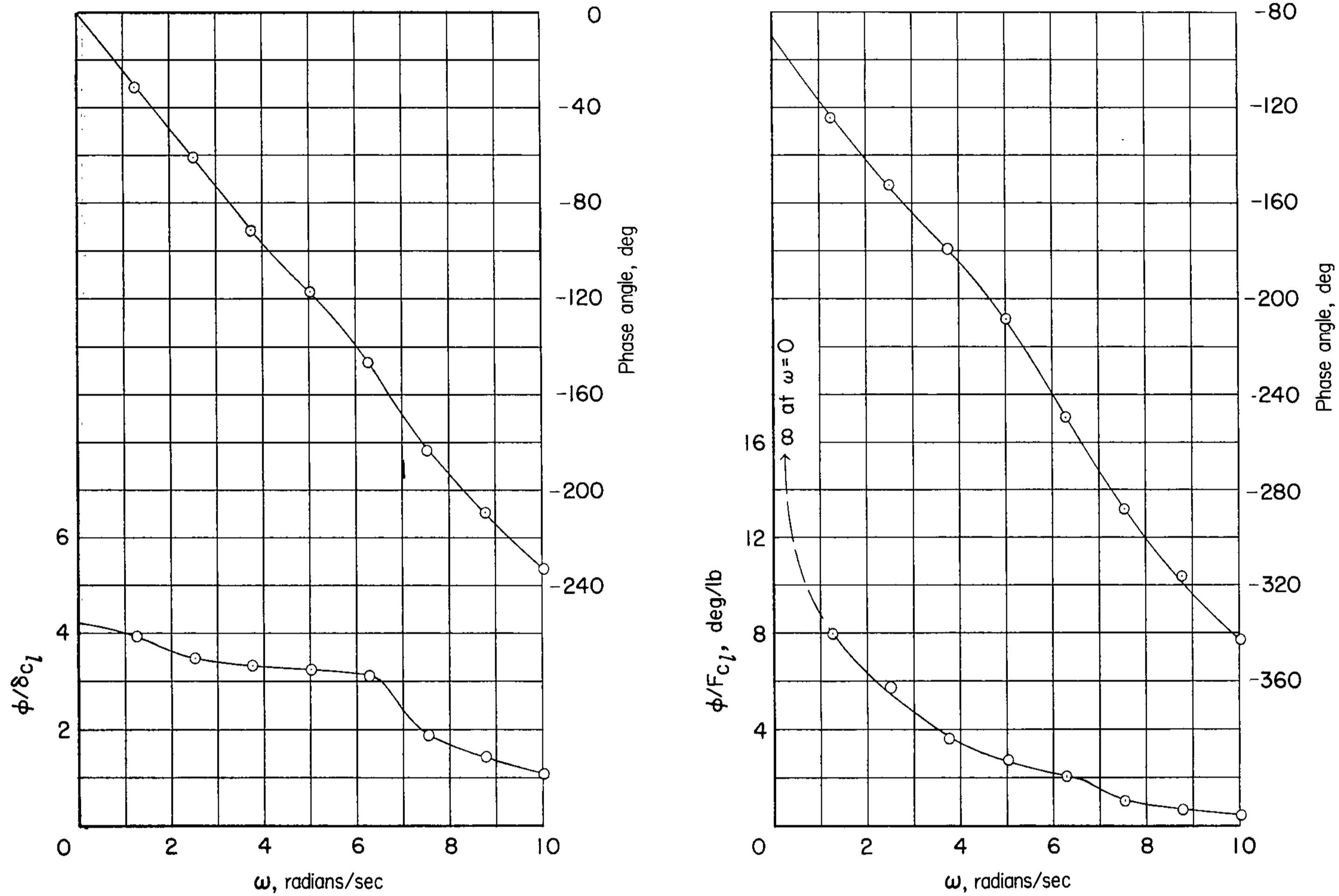
Figure 14.- Continued.



(c) Power approach condition, power for level flight, V_i = 125 knots, h_p = 5,000 feet, K_{ϕ} = 14.3 volts/radian, $K_{\dot{\phi}}$ = 6.5 volts/radian/sec, K_{f_a} = 7.0 volts/radian.

(d) Clean condition, power for level flight, M = 0.6, h_p = 5,000 feet, K_{ϕ} = 14.3 volts/radian, $K_{\dot{\phi}}$ = 6.5 volts/radian/sec, K_{f_a} = 7.0 volts/radian.

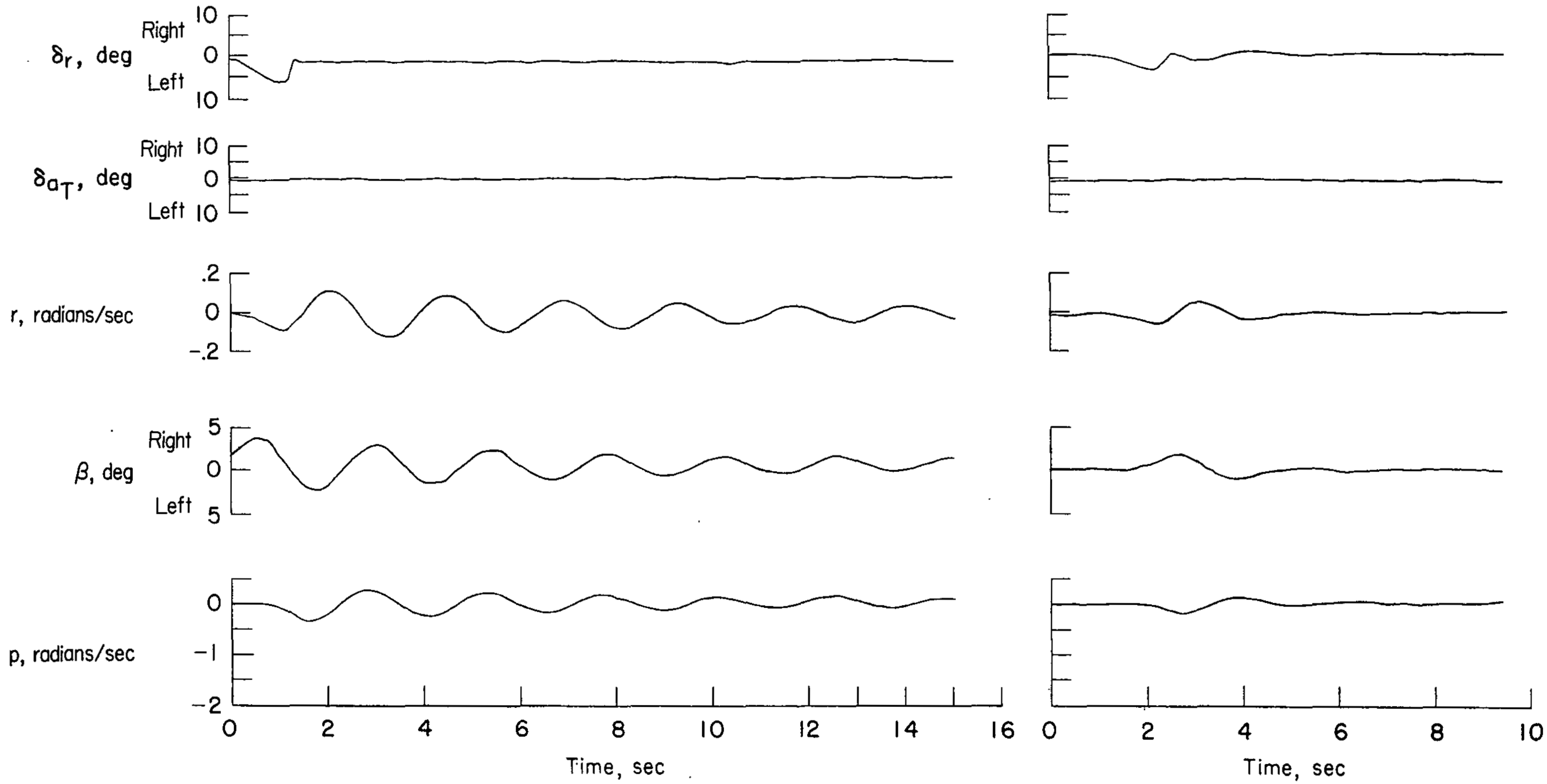
Figure 14.- Concluded.



(a) ϕ/δ_{c_l}, h_p = 5,000 feet.

(b) ϕ/F_{c_l}, h_p = 5,000 feet.

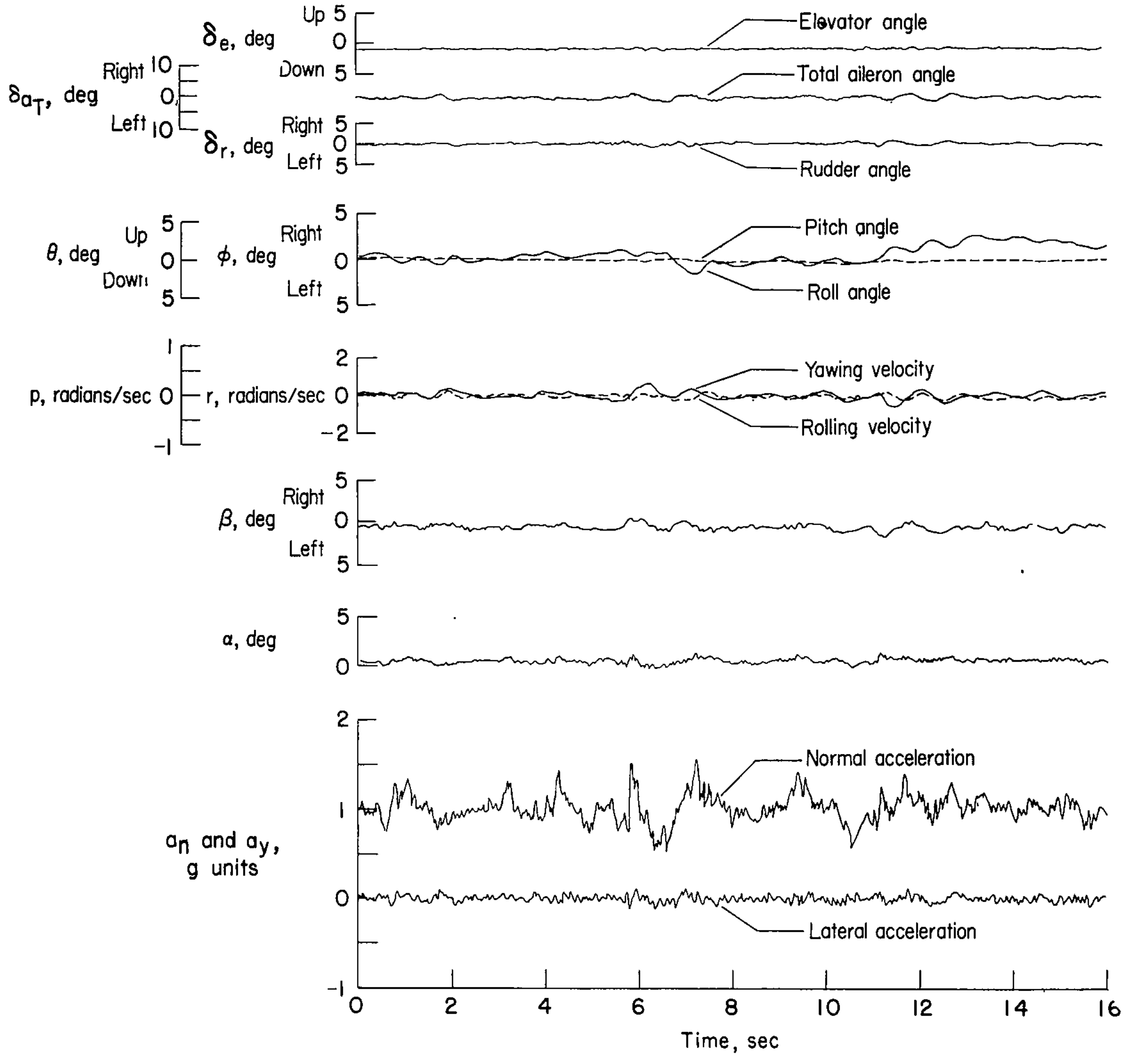
Figure 15.- Frequency-response characteristics in roll of the airplane—automatic-pilot combination with damper feel system. M = 0.6, K_ϕ = 14.3 volts/radian, $K_{\dot{\phi}}$ = 4.0 volts/radian/sec, K_{f_a} = 7.0 volts/radian.



(a) Basic airplane.

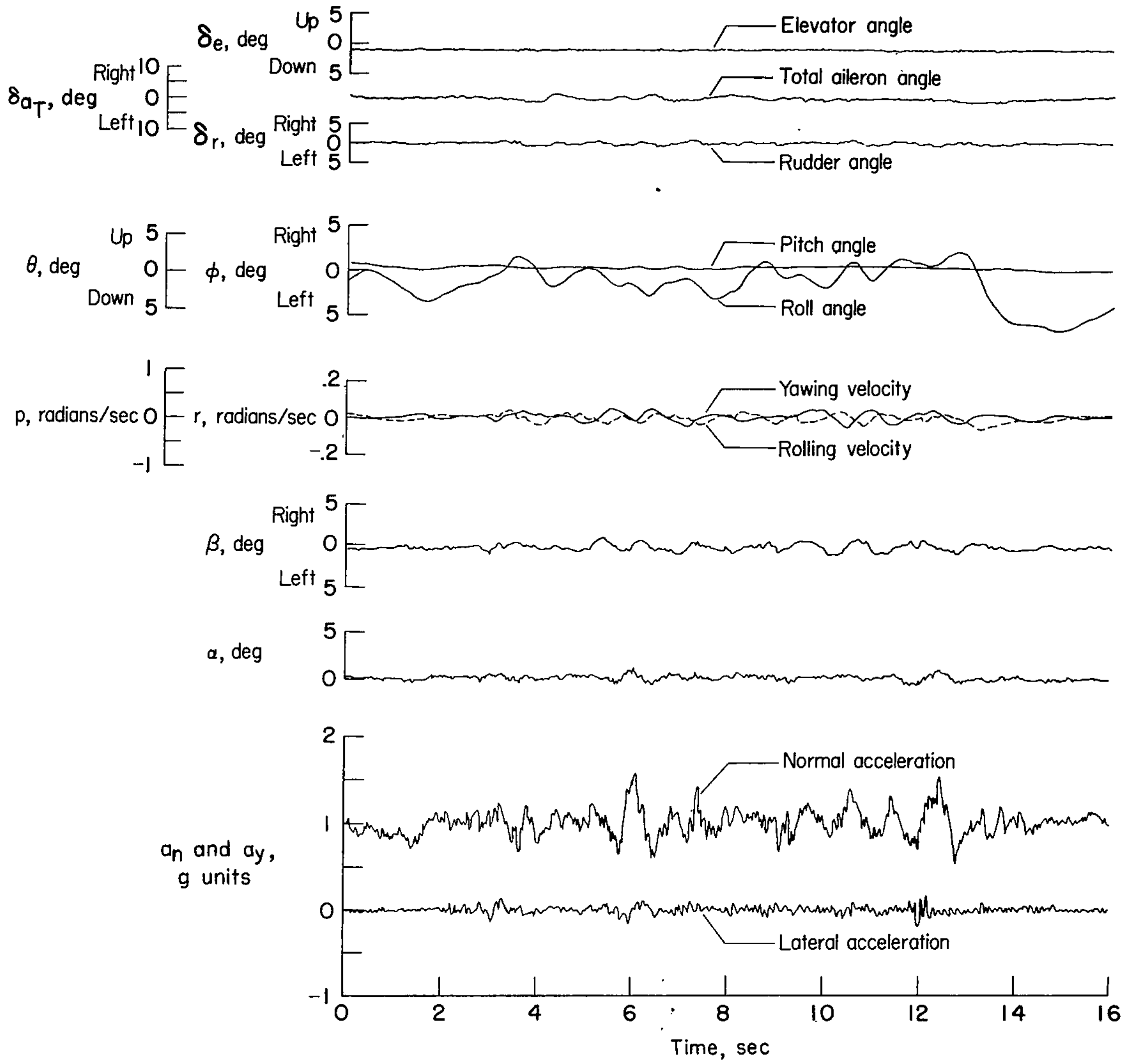
(b) Airplane with rudder channel in operation, $K_{\dot{\psi}} = 20.1$ volts/radian/sec, $K_{\beta} = 16.4$ volts/g, $K_{f_r} = 5.7$ volts/radian.

Figure 16.- Time histories of rudder kicks at $M = 0.6$ and $h_p = 30{,}000$ feet.



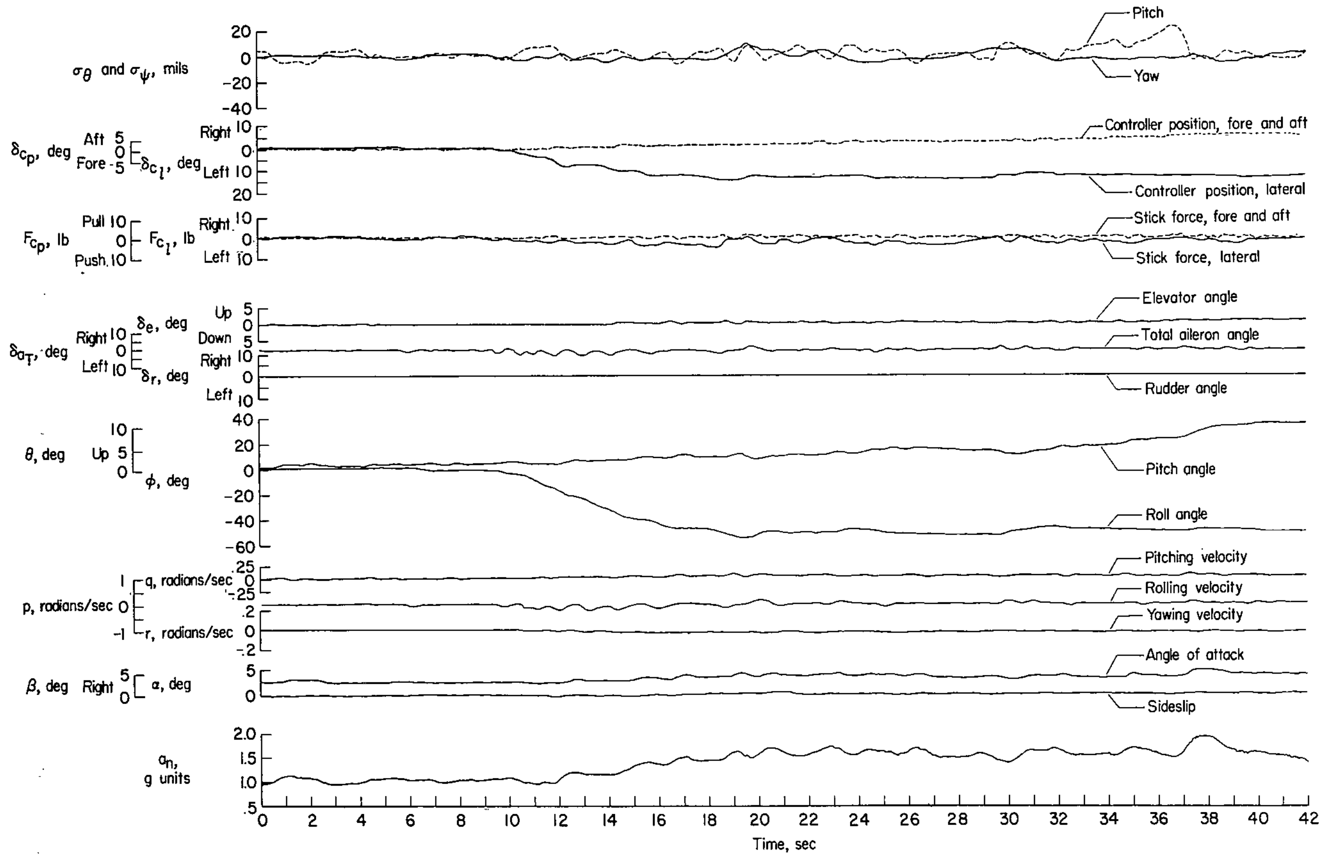
(a) Automatic pilot, $K_\theta = 15.5$ volts/radian, $K_{\dot{\theta}} = 11.6$ volts/radian/sec, $K_{f_e} = 7.0$ volts/radian, $K_\phi = 14.3$ volts/radian, $K_{\dot{\phi}} = 6.5$ volts/radian/sec, $K_\psi = 23.0$ volts/radian, $K_{f_a} = 7.0$ volts/radian, $K_{\dot{\psi}} = 20.1$ volts/radian/sec, $K_\beta = 16.4$ volts/g, $K_{f_r} = 5.7$ volts/radian.

Figure 17.- Time histories of straight and level flight in rough air. $M = 0.6$, $h_p = 5,000$ feet.



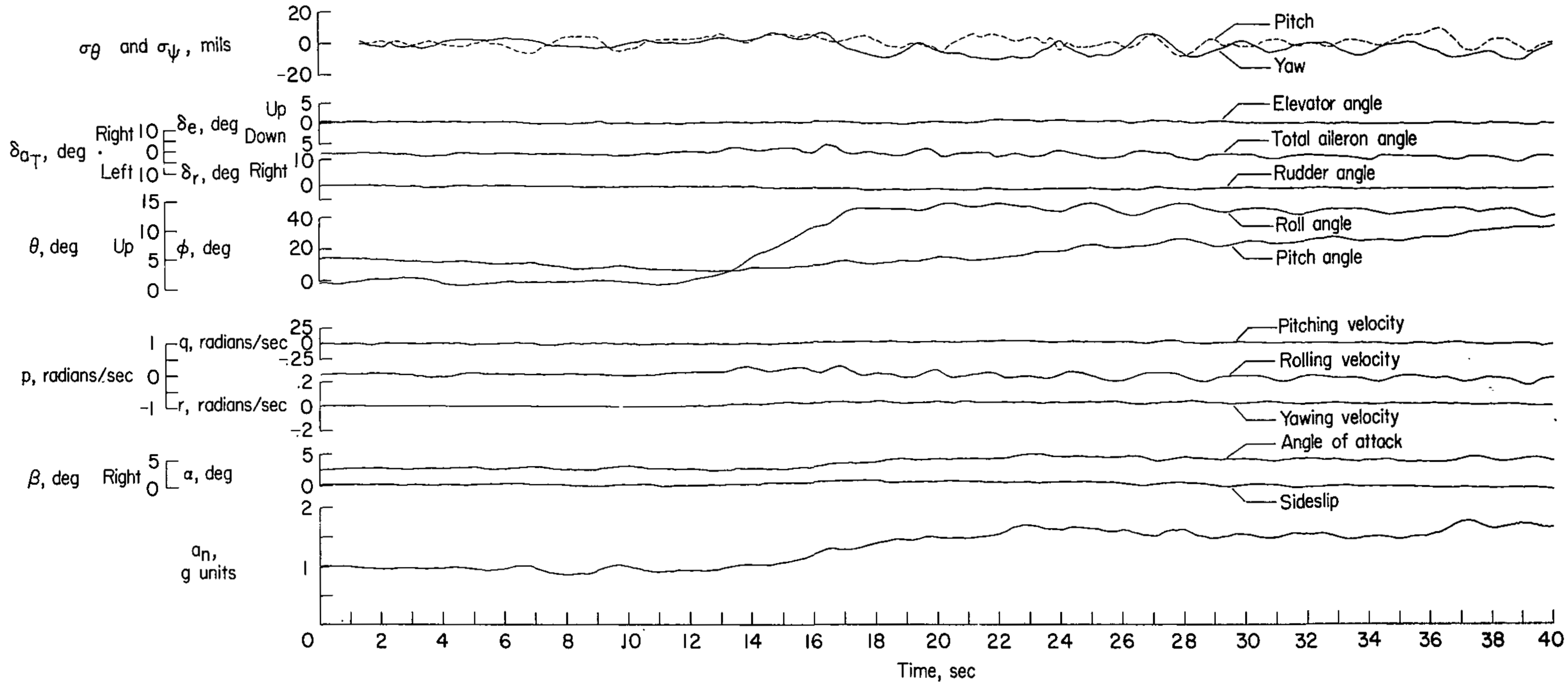
(b) Conventional control system with rudder channel in operation.

Figure 17.- Concluded.



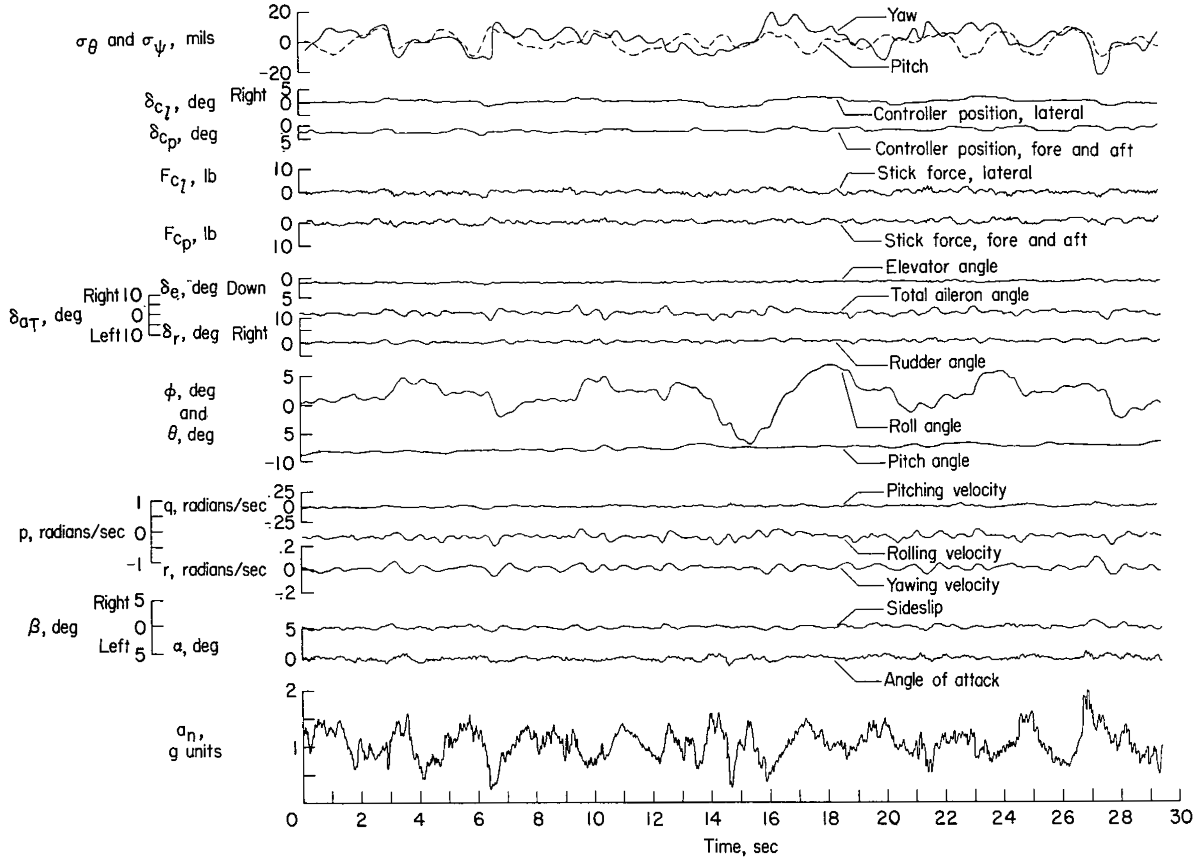
(a) Automatic pilot, $K_\theta = 15.5$ volts/radian, $K_{\dot{\theta}} = 9.2$ volts/radian/sec, $K_{f_e} = 3.5$ volts/radian, $K_\phi = 14.3$ volts/radian, $K_{\dot{\phi}} = 16.7$ volts/radian/sec, $K_{f_a} = 7.0$ volts/radians, $K_{\dot{\psi}} = 20.1$ volts/radian/sec, $K_\beta = 16.4$ volts/g, $K_{f_r} = 5.7$ volts/radian.

Figure 18.- Time histories of tracking runs in turns. $M = 0.6$, $h_p = 30{,}000$ feet.



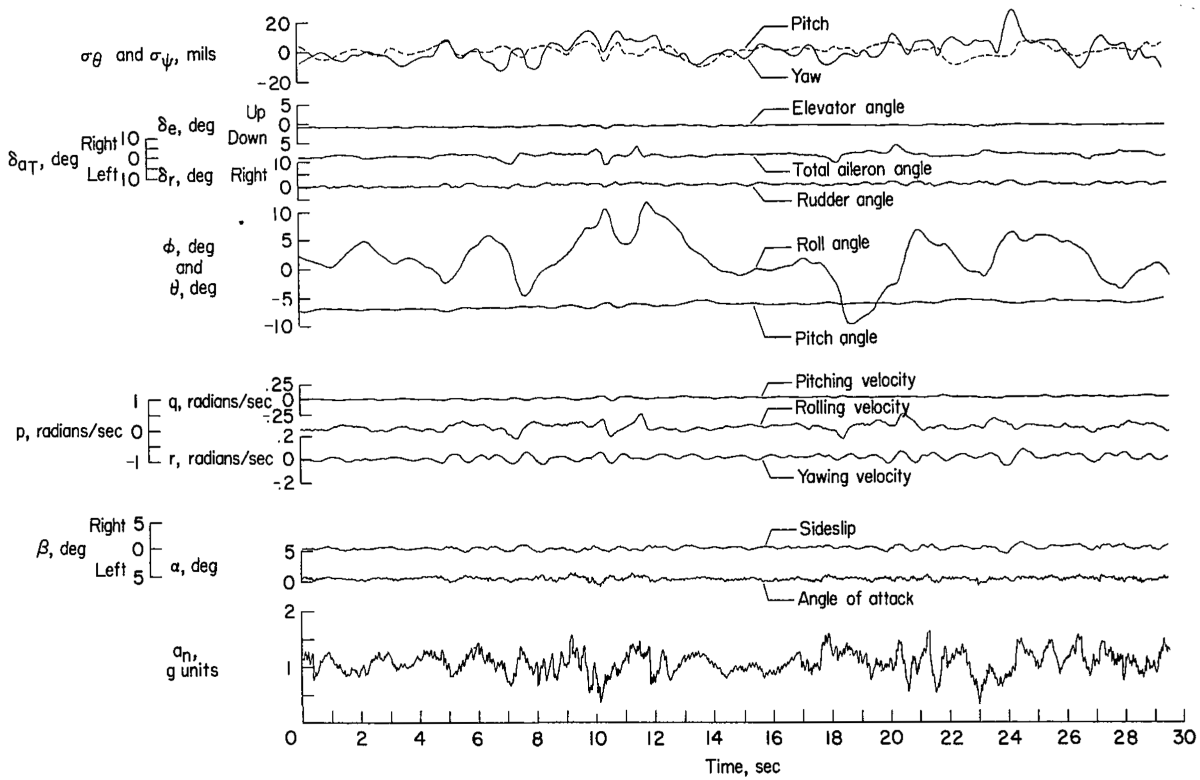
(b) Conventional control system with rudder channel in operation.

Figure 18.- Concluded.



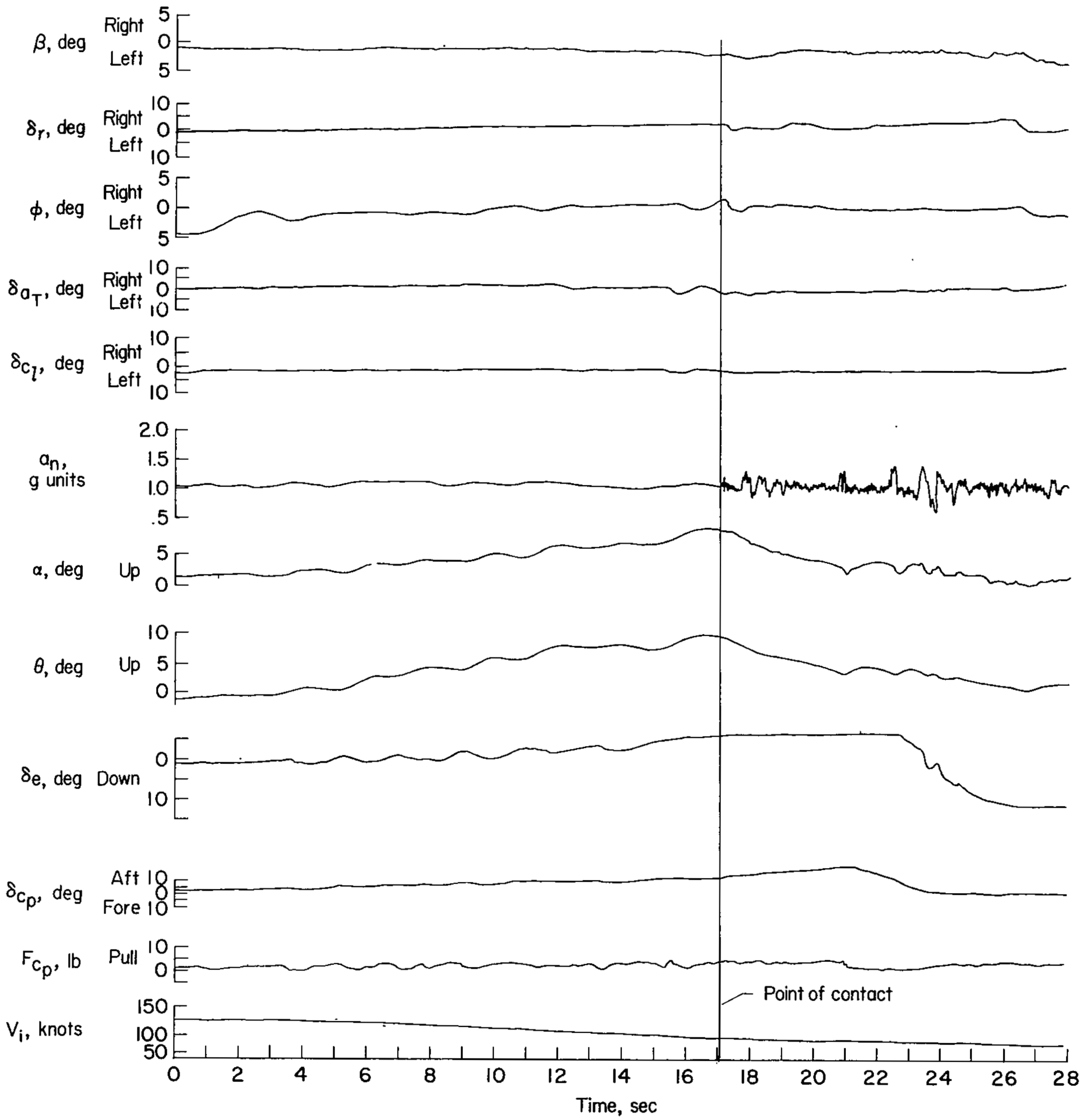
(a) Automatic pilot, K_θ = 15.5 volts/radian, $K_{\dot{\theta}}$ = 11.6 volts/radian/sec, K_{f_e} = 7.0 volts/radian, K_ϕ = 14.3 volts/radian, $K_{\dot{\phi}}$ = 6.5 volts/radian/sec, K_{f_a} = 7.0 volts/radian, $K_{\dot{\psi}}$ = 20.1 volts/radian/sec, K_β = 16.4 volts/g, K_{f_r} = 5.7 volts/radian.

Figure 19.- Time histories of strafing runs in rough air. M = 0.6, h_p = 3,000 feet to 1,000 feet.



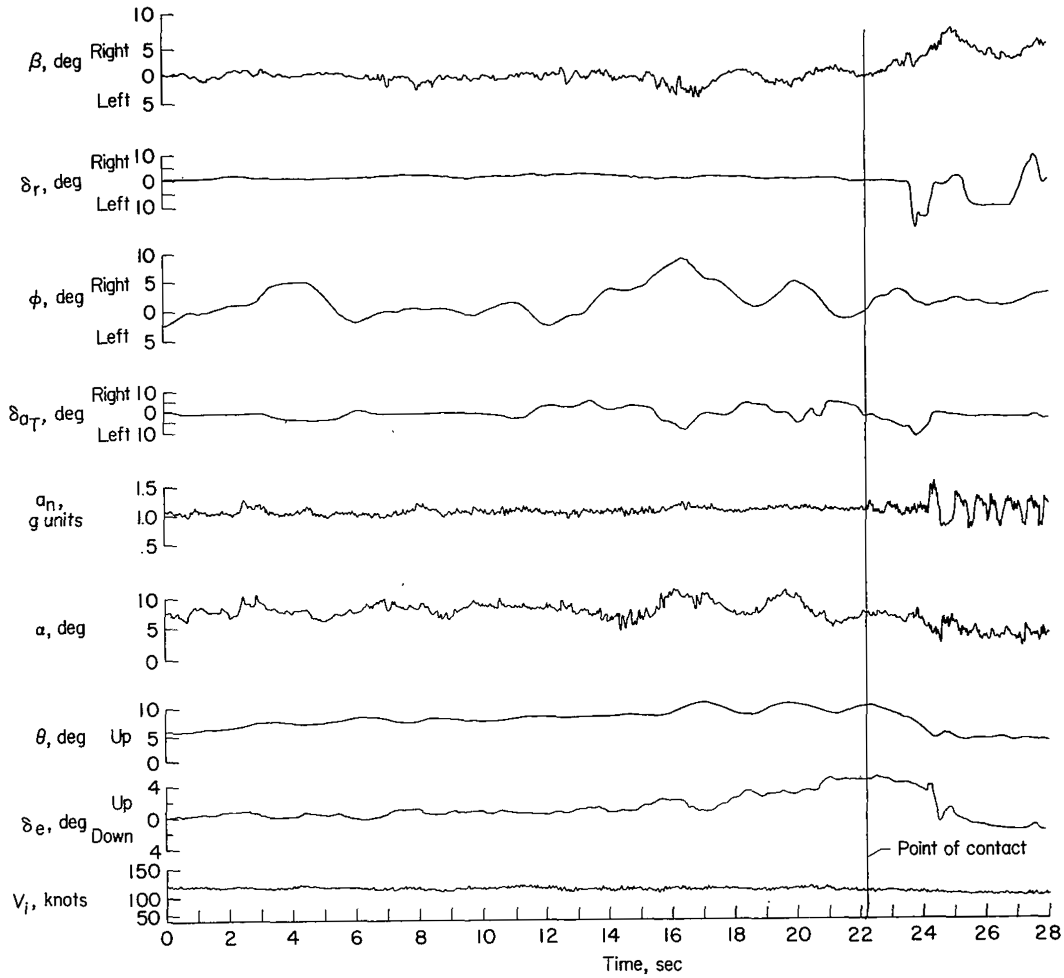
(b) Conventional control system with rudder channel in operation.

Figure 19.- Concluded.



(a) Automatic pilot, K_θ = 15.5 volts/radian,
$K_{\dot{\theta}}$ = 16.8 volts/radian/sec, K_{f_e} = 7.0 volts/radian,
K_ϕ = 14.3 volts/radian, $K_{\dot{\phi}}$ = 6.5 volts/radian/sec,
K_{f_a} = 7.0 volts/radians, $K_{\dot{\psi}}$ = 20.1 volts/radian/sec,
K_β = 16.4 volts/g, K_{f_r} = 5.7 volts/radian.

Figure 20.- Time histories of landings.



(b) Conventional control system.

Figure 20.- Concluded.

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